NASA/CR-2003-212652

Engineering Feasibility and Trade Studies for the NASA/VSGC MicroMaps Space Mission

Ossama O. Abdelkhalik, Bassem Nairouz, Timothy Weaver, and Brett Newman
Old Dominion University, Norfolk, Virginia

September 2003

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Space Administration

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Prepared for Langley Research Center
under Purchase Order L-16151

September 2003

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Section I
Introduction

This report describes the activities and accomplishments conducted under contract VSGC-521991 with the Virginia Space Grant Consortium (VSGC), and indirectly with the National Aeronautics and Space Administration Langley Research Center (NASA LaRC). Subject matter is engineering feasibility and trade studies for the NASA/VSGC MicroMaps Space Mission. Specific components addressed within the overall NASA/VSGC MicroMaps Mission effort include 1) assess and recommend MicroMaps instrument space basing platform options, 2) survey and catalog near term launch vehicle opportunities for MicroMaps space access, 3) investigate and highlight MicroMaps instrument on-orbit thermal control requirements and solutions to maintain scientific measurement integrity, and 4) study feasibility of adding an imaging component to the MicroMaps instrument system for scientific or educational purposes. Components 1 and 2, and some aspects of component 4, are conducted by Aerospace Engineering Department students, and associated findings are documented in this report. Components 3 and 4 are conducted by Mechanical Engineering Department students. Their findings are documented in a separate report.[1]

Contract activities are of a preliminary first-order engineering feasibility assessment nature conducted on a short duration schedule with limited resources and input information. Results and recommendations from these activities are envisioned to support future MicroMaps Mission design decisions regarding policy and program down select options leading to more advanced and mature phases. Quantifying the merits and/or deficiencies of the options, in terms of facilitating scientific objectives, cost and complexity, reliability and robustness, and sizing and requirements, should be an integral part of the activities. Project objectives are three fold: 1) to conduct studies in direct support of MicroMaps Mission decision making, 2) to develop student-based technical capabilities in the area of engineering support for space systems, and 3) to enhance student educational training with applied experience in critical areas of need such as space systems.

Natural and/or human surface activities, such as biomass burning or industrial processing, release significant concentrations of gaseous by-products such as carbon monoxide (CO) into the atmosphere. Trace CO gases can be transported by natural phenomenon over great distances and altitudes, and undergo mixing and chemical reaction with other natural elements like oxygen-hydrogen radicals (e.g., OH). Reduction of upper atmospheric OH content may adversely affect the natural removal of undesirable greenhouse gases such as methane (CH_4). Further, CH_4 is tightly coupled to the dynamic life cycle of atmosphere ozone (O_3) composition. These mechanisms may have significant influence on the Earth's greenhouse effect and global climate trends.[2,3] At this time, these large scale dynamic processes are not well understood. Further, scientific data such as CO spatial and temporal distributions to be used as inputs for global atmospheric and climate prediction models is severely lacking. A critical need for expanded atmospheric CO databases exists so that accurate scientific predictions can be undertaken and reported to appropriate governing political bodies making large scale environmental policy and regulation.

MicroMaps is an existing NASA owned gas filter radiometer instrument with 3 deg field of view designed for space-based nadir measurement of atmospheric CO vertical profiles in the 4.67 μm wavelength.[4] The instrument was part of an overall scientific mission to be flown on the latter of the two Lewis and Clark spacecraft. Unfortunately, this mission was canceled leaving the completed instrument without access to the space environment.[5,6] Currently, the instrument is in storage with nominal but dated performance capabilities. MicroMaps hardware has high potential for filling a critical scientific need, thus motivating concept studies for new and innovative scientific spaceflight missions that would leverage the MicroMaps heritage and investment, and contribute to new CO distribution data to be used in global-scale atmosphere and climate modeling and prediction. Conceptual studies should encompass a broad spectrum of topics from launch options and platform design requirements to instrument operations and scientific post-processing of the measurement data. Consideration of options for instrument refurbishment and/or enhancement with low cost retrofit upgrades is also needed. Only a subset of these topics are addressed in this limited scope project, as outlined below.

Section II describes analysis and synthesis methodology for the MicroMaps Space Mission. A generalized mission planning process applicable to any space mission is described and offered. This process is also discussed in the context of the MicroMaps Space Mission. However, limited resources constrain application of this process only to selected areas of the MicroMaps Space Mission. Emphasis is also given to development of the requirement flow down relationships where science objectives, instrument specifications, environment factors, and resource reserves are used to formulate requirements on such aspects as orbit design, platform selection, and subsystem sizing and definition. Such relationships can be used to expose critical factors which impact the overall system design. Associated insight may be more valuable for program decision making than specific subsystem definition and sizing studies. Because the flight trajectory impacts so many other factors and subsystems, orbit selection is also given special attention in this section.

Section III describes subsystem studies and detailed requirements development for the MicroMaps orbital platform option consisting of a small dedicated spacecraft with a single purpose mission. Development of this vehicle is envisioned to be primarily an in-house construction and fabrication effort involving the NASA/VSGC student team where feasible, supplemented with integration activities of purchased components. Spacecraft subsystems addressed in these studies include attitude sensing and control, orbital adjustment and maintenance propulsion, electrical power generation and storage, vehicle-ground communication and telemetery, and Earth observation camera. From all potentially necessary vehicle systems, this subsystem list was chosen based on its perceived mission criticality and an attempt to match subsystem discipline with available student team member technical capabilities and interests. Emphasis is given to definition and sizing of specific hardware components that will meet mission objectives as defined at this time. Because the mission requirements are not fully defined at this time, in some instances assumptions will be noted and invoked.

Section IV describes key issues associated with the MicroMaps orbital platform option consisting of the International Space Station, a large space structure with multi-purpose functions. With this option, interfacing instrument and support subsystems with the International Space Station infrastructure to ensure scientific objectives are satisfied would be the primary engineering challenge for the NASA/VSGC student team. Key issues related to the International Space Station addressed here include Earth surface coverage, attitude and vibration transients, and active pointing and vibration isolation systems. From all potentially significant design issues, this list of key issues was chosen based on its perceived importance to mission design complexity and cost implications, and an attempt to match technical issues with available student team member expertise and interests. Emphasis is given to characterization of losses in scientific content and quality due to International Space Station environmental factors imposed on the instrument such as orbital geometry or motion transients, or the cost/complexity required to maintain scientific mission integrity under these environmental platform constraints.

Section V describes potential launch opportunities for gaining assess to orbit for the MicroMaps instrument, regardless of the orbital platform option chosen. Due to program resource limitations, low cost ride sharing or piggyback arrangements on unrelated but compatible space launch missions is the only feasible option for gaining access to orbit. Emphasis is given to domestic government and/or commercial low Earth orbit launches with a schedule of approximately 3-5 years off from the present time. Finding an ample basis of planned Earth space missions beyond the 1 year near term focus proved to be difficult, but several candidate missions were identified. These potential launch opportunities are then analyzed and assessed for applicability to the MicroMaps Mission using criteria such as orbital geometry insertion, inertial and geometric launch constraints, vibration, thermal and acoustic launch environments, payload launch vehicle interfacing, schedule availability, cost, and willingness to cooperate.

Section II
Mission Analysis and Synthesis

A. Generalized Mission Planning

The MicroMaps Space Mission is currently in the early stages of mission analysis and synthesis. Working in this phase requires careful integration and coordination of developers, sponsors, operators, and users (or customers) all as a team in order to extract maximum performance from the mission at minimum cost. In this subsection, generalized steps for space mission analysis are briefly written down where applicable. These steps are also related to and discussed in terms of specific aspects of the MicroMaps Mission. Because the main payload is already developed (i.e., the MicroMaps science instrument), some steps may not be applicable or are already completed in the generalized methodology. Generalized space mission analysis and synthesis methodology may go through the following steps, which are further detailed below.[7,8]

- Define Mission Objectives and Goals
- Define Mission Constraints and Requirements
- Define Mission Concepts and Architectures
- Define Mission Components and Elements
- Characterize Mission Concepts and Components
- Evaluate Mission Concepts and Components
- Select Preferred Mission Concept and Components
- Refine Mission Objectives, Constraints and Requirements

<u>Define Mission Objectives and Goals</u>

Mission objectives and goals are usually the genesis of any space mission and arise out of a need to explore or exploit space for scientific, commercial, or other purposes. Conception of most space missions start with well defined needs. Thus, mission objectives are easily identified in most cases and formalized in a mission statement. Objectives may be divided into primary objectives and secondary objectives that can be met by the defined set of equipment, and additional objectives that may demand more equipment. Objectives may be modified slightly or not at all during mission analysis and synthesis as various capabilities and technologies are weighed against associated costs and complexity in various competing architectures.

In the MicroMaps application, mission objectives are rather well defined. The overriding objective is to collect Earth atmospheric CO spatial and temporal distributions by measurements taken form space of sufficient quality and quantity to be used as inputs for global atmospheric and climate prediction models in scientific investigations. The spatial and temporal distributions should have the resolution to address global and regional effects, as well as monthly, seasonal and annual variations. Secondary objectives are to develop and enhance student technical capabilities and educational experience in the areas of space systems. Mission objectives can also be summarized as

Main Objectives:

1. Define seasonal and inter annual evolution of the strengths of the CO sources and sinks,
2. Enhance the temporal and spatial resolution of other space-based CO remote sensors such as MOPITT and TES,
3. Provide complementary measures to, and extend the context and scope of, airborne measurement campaigns such as GTE or EOS validation missions,

Additional Objectives:

1. Obtain a reasonable resolution image associated with each measurement location.

Define Mission Constraints and Requirements

From the objectives and any imposed constraints, functional and operational requirements are to be formulated and quantified to the extent possible at this early stage. Before doing this task, the necessary input information must be collected. Mission objectives are available from the previous step. All relevant constraints should be identified and defined here. These

MOPITT: Measurement Of Pollution In The Troposphere
TES: Tropospheric Emission Spectrometer
GTE: Global Tropospheric Experiment
EOS: Earth Observing System

constraints can originate from a variety of sources such as physical, environmental, technological, regulatory, financial, or social factors. For example, constraints could arise from a time schedule, budget shortage, hardware limitation, or space environment, to name just a few. Once this input information is at hand, the requirements should be defined. For example, from the objectives and constraints one could potentially derive an upper bound on the required pointing accuracy of a satellite, an acceptable family of orbital geometries, or the minimum number of required satellites. These requirements will be iterated in further analysis and synthesis steps to follow.

In the MicroMaps application, a set of applicable mission constraints can be partially defined at this time. These constraints can be classified into sources including Science, Instrument, Environment, Launch, Resources, and Technology. This information is documented in Section II-B and includes factors such as pointing accuracy imposed by scientific data fidelity, telemetry rates imposed by the MicroMaps instrument, atmospheric density and Van Allen radiation altitude windows imposed by the environment, vibrational transmissions imposed by the launch conditions, and project funding imposed by available resources. Finally, the constraint and objective information can be translated into requirements for acceptable orbital geometries and associated atmospheric measurements, acceptable attitude sensing and pointing and associated hardware components, acceptable data storage and transmission capability and associated hardware, for example.

Define Mission Concepts and Architectures

In this step, various mission concepts and architectures are collected and defined. This effort essentially defines a set of competing options and how they will function and operate in practice to achieve the mission objectives. This set of concepts should be populated with significantly different mission implementation strategies to sufficiently cover a large design space. However, the more options that are considered, the more effort that is required in latter mission analysis and synthesis steps. This step can be thought of as a focused brainstorming process. This step is where engineering creativity and ingenuity can play a significant role.

In the MicroMaps application, high level concept definitions might address the following options that will be subject to trades associated with advantages and disadvantages.

- Distribution: Single Platform, Bi Platform, Multi Platform
- Trajectory: Low vs. High Inclination, Free Drift vs. Orbit Maintenance
- Processing: Processing on Ground, Hybrid Strategy, Processing in Orbit
- Telemetry: Direct To Ground Station, Space Network Relay, Amateur Radio Station
- Operation: Highly Autonomous, Mixed Strategy, Significant Human Involvement
- Insertion: Spring Loaded, Compressed Jet, Grapple Release
- Fabrication: In House, Purchase and Integrate, Contract Out

Define Mission Components and Elements

For each mission concept from the previous step, various mission components and elements that make up each concept are proposed and defined. Each mission concept can be thought of as a specific strategy to achieve system functions, while the components are interpreted as specific subsystems, or subsystems of subsystems, used to implement and mechanize these strategies. This effort also defines a set of lower level competing options for achieving mission objectives. This set of components should also be populated with different subsystem approaches to cover a large design space, while simultaneously avoiding a computationally intractable set of options. Engineering creativity and ingenuity can play a significant role here as well.

In the MicroMaps application, low level component definitions might address the following options that will be subject to trades associated with advantages and disadvantages.

- Platform: Dedicated Spacecraft, International Space Station, High-Altitude Aircraft
- Orbit: Sun-Synchronous, Earth-Synchronous, A-Synchronous
- Power: Solar, Fuel Cell, Battery Reserve, Hybrid
- Telemetry: Single Ground Station vs. Multi Ground Stations,
  High Storage - Low Rate Transmitter - Fixed Antenna vs.

Low Storage - High Rate Transmitter - Slewed Antenna

- Propulsion: None, Electrothermal, Electrostatic, Electromagnetic
- Control: Magnetic Torque, Momentum Wheel, Inertial/Satellite Navigation, Star Track
- Launch: Space Shuttle, Expendable Vehicle, Vertical vs. Air Drop

Characterize Mission Concepts and Components

Once the mission concepts and components are defined, their capabilities and limitations must be characterized and quantified for later evaluation and assessment. This step typically involves collecting relevant performance information for each electrical-mechanical hardware component, for example, and describing each component with an appropriate engineering math model. In addition to this component characterization, modeling of the interaction and coupling of the components to build up the whole system may be required. Proper inclusion of constraints into these models is also required. These characterization efforts can be conducted with varying degrees of fidelity and precision, depending upon factors such as constraining schedules, available resources, desired insights, or required accuracies.

Evaluate Mission Concepts and Components

In this step, utility of various mission concepts and components are evaluated and assessed. A set of formal criteria or metrics are formulated to judge and rank the available options. This set of criteria should encompass factors deemed important to mission success by the mission designer such as performance, complexity, reliability, cost, and risk, to name just a few. These various factors can be assigned relative weights to emphasize specific goals. Quantification of some factors may pose a difficult challenge. These criteria are then applied to the competing mission options. Benchmarking how well each option is meeting both the requirements and objectives under the imposed constraints, as a function of cost or key system design choices, is the desired end result. The process should ultimately provide the decision maker with a single chart of potential performance vs. required cost, from which the best or optimum mission design option is extracted. An important by-product of this process is identification of critical constraints and requirements, or key system parameters or drivers, that strongly influence evaluation criteria. Such insight is invaluable to mission designers and program managers.

Select Preferred Mission Concept and Components

Using results from the previous step, a preferred of favored mission concept and associated components is down selected for further planning or implementation. In some cases, a clear cut choice is obvious while in other cases it is not so clear. In these latter situations, and if resources allow, two or more options are carried along to more advanced stages of mission analysis and synthesis where the preferred option may become apparent. This approach also reduces risk to unexpected technical difficulties and their solution that were not exposed in preliminary analysis efforts. If two or more mission options are truly competitive, the designer may have to resort to engineering intuition or heritage legacy, for example, to make the final down select decision.

Refine Mission Objectives, Constraints and Requirements

Mission planning may lead to situations where objectives are overly aggressive or constraints are exceedingly harsh. In such situations, several or all requirements may not be achievable and mission objectives, constraints, and requirements may need refinement. Some constraints are hard constraints not subject to the authority of the mission planner. Other constraints, however, may be of an adjustable nature that can be relaxed. In addition, the designer may chose to soften the objectives. Insight concerning the critical constraints and requirements, and their functional dependency on key system drivers or parameters, is used to support these changes. After making these decisions, the designer reformulates the mission requirements imposed by the new objectives and constraints, and previous steps are revisited. Pursuing this refinement process turns the mission analysis and synthesis methodology into an iterative process. This step is not always required.

B. Requirement Flow Down Relationships

In this subsection, development of the requirement flow down relationships for the MicroMaps Space Mission are addressed to the extent that contract resources allow, and to the extent that available information allows, in this early stage of mission analysis and synthesis. In this process, objectives and constraints such as science goals, instrument specifications, environment constraints, and resource reserves are used to formulate requirements on such mission aspects as orbit design, platform selection, and subsystem sizing and definition. Formulation of the most significant mechanisms and mappings of objectives and constraints into requirements related to orbit design and selected subsystem definition (for a small dedicated spacecraft platform) will be emphasized here. Details of these relationships are presented in Section III. Such relationships can be used to expose critical factors which impact the overall system design. Associated insight may be more valuable for program decision making than specific subsystem definition and sizing studies.

Figure II-B.1 illustrates the basic components involved in the flow down relationships. At the top level, objectives and constraints from factors such as Science, Instrument, Environment, Launch, Resources, and Technology are shown. These factors represent known objective and constraint information, and serve as input for the formulation process of flow down relationships. Other input factors can be incorporated into Figure II-B.1 as they become known. At the bottom level, requirements on spacecraft subsystems related to Control, Propulsion, Electrical, Telemetry, and Camera are shown. These components represent unknown requirement information that serves as output from the formulation process for flow down relationships. Other output factors could be incorporated into Figure II-B.1, if desired. An intermediate level associated with Orbit Geometry is also shown in Figure II-B.1. Requirements for Orbit Geometry are influenced by many objectives and constraints. In turn, the orbit characteristics influence many spacecraft subsystem requirements. Because Orbit Geometry receives and transmits many key flow down relationships, it is given special consideration.

Top Level

Science	Instrument	Environment	Launch	Resources	Technology
Latitude Coverage	Size	Atmospheric Density	Size	Financial	Performance Limits
Coverage Rate	Field of View	Gravity Disturbance	Vibration	Hardware	Reliability
Attitude Accuracy	Data Rate/Storage	Solar Activity	Acoustic	Facilities	Capability vs. Cost
Attitude Knowledge	Power Usage	Radiation Intensity	Thermal	Testing	
Position Accuracy	Heat Sensitivity	Magnetic Intensity	Interface		
Position Knowledge	Disturbances	Solar Intensity	Safety		
	Life Rating	Cloud Statistics	Available Orbit		
		Ground Station			

Intermediate Level

Orbit Geometry
Altitude
Inclination
Synchronicity
Stability

Bottom Level

Control	Propulsion	Electrical	Telemetry	Camera
Angular Detection	Thrust Level	Power Generation	Data Rate	Aperture Diameter
Moment Generation	Impulse Total	Energy Storage	Data Storage	Focal Length
Attitude Authority	Sizing & Definition	Surface Area	Downlink Power	Detectors
Linear Detection	Cost & Complexity	Distribution	Antenna Gain	Pixel Width
Force Generation		Sizing & Definition	Uplink Sensitivity	Data Rate
Position Authority		Cost & Complexity	Sizing & Definition	Sizing & Definition
Attitude Maneuver			Cost & Complexity	Cost & Complexity
Sizing & Definition				
Cost & Complexity				

Figure II-B.1 Flow Down Relationship Components

As a starting point, all known information relating to mission objectives and constraints are collected. At this early stage of mission analysis and synthesis, the following partial list of information was collected. Science and Instrument data originates primarily from Reference 4 and discussions with Dr. Vickie Connors (NASA LaRC) and Dr. Henry Reichle (NASA LaRC Retired). Environment data originates from known facts documented in many texts such as References 9-10. With no specific launch opportunity identified at this time, the Small Spacecraft Technology Initiative (SSTI) design requirements for ascent conditions are interpreted as actual ascent conditions.[4] Note access to the NASA Spaceflight Tracking and Data Network (STDN) is assumed here. This information is tentative and could evolve as the mission design proceeds.

<u>Science</u>
- Coverage of Major CO Sources and Sinks: Latitudes From 0 deg to Beyond 75 deg
- Temporal Resolution in CO: Complete Coverage Every 30 days
- Spatial Resolution in CO: 5 deg Longitude by 5 deg Latitude
- Pointing Knowledge for Data Fidelity: ± 0.5 deg
- Positional Knowledge for Data Fidelity: ± 25 km
- Pointing Accuracy for Data Fidelity: ± 5 deg Nadir (Ref. 4 Lists ± 2.5 deg Nadir)
- Pointing/Positional Update: 0.1 Hz

<u>Instrument</u>
- Life Rating: 3 years
- Dimensions: 6 in High, 8.25 in Wide, 13.75 in Deep
- Mass: 6.4 kg
- Inertias: $I_{xx} = 0.049$, $I_{yy} = 0.047$, $I_{zz} = 0.030$, $I_{xy} \approx I_{yz} \approx I_{zx} \approx 0$ kg m^2
- Power Consumption: 24 W
- Input Voltages: +15, -15, +5 V
- Communication Interface: RS 422 with XMODEM
- Data Sampling MicroProcessors: Hitachi 6303
- Data Processing MicroProcessor: RHC 3000
- Data Rate: 288.7 bit/s Uncompressed, 40 bit/s = 0.432 Mbyte/day Compressed
- Data Storage Buffer: FIFO Circular 0.432 Mbyte (1 Downlink per Day)
- Field of View: ± 1.5 deg Cone
- Circular Footprint from Low Earth Orbit: 25 km Diameter
- Sensitive Wavelength: 4.67 μm
- Detector Temperature: 0 to 25 deg C
- Chopper Max Momentum Disturbance: 0.05 lbf ft s
- Chopper Inertia Imbalance: ± 18 mg at 2 in Radius
- Chopper Frequency: 2,000 rpm

- Calibration Assembly Max Torque Disturbance: 0.004 Nm every 2.5 s
- Calibration Assembly Frequency: 30 min Cycle per day
- Radiation Exposure: 10 krads Total, 30 MeV Upset Free, 100 MeV Latchup Free
- Magnetic Dipole: 0.2 Am^2 Induced from 21 Am^2 Exposure, 0.01 Am^2 Residual

<u>Environment</u>

- Gravitational Disturbances: J_2 Oblate Earth Model
- Atmospheric Density:

Solar Max	Solar Min	Orbit
3.39×10^{-10} kg/m^3	1.69×10^{-10} kg/m^3	200 km
2.56×10^{-11} kg/m^3	1.28×10^{-11} kg/m^3	300 km
7.93×10^{-12} kg/m^3	2.36×10^{-12} kg/m^3	400 km
2.44×10^{-12} kg/m^3	3.26×10^{-13} kg/m^3	500 km
8.62×10^{-13} kg/m^3	5.81×10^{-14} kg/m^3	600 km
3.67×10^{-13} kg/m^3	1.61×10^{-14} kg/m^3	700 km

- Radiation Intensity: ≈0 krads Every 10 years at 600 km, Common Shielding
  3 krads Every 10 years at 800 km, Common Shielding
  27 krads Every 10 years at 1,000 km, Common Shielding
- Magnetic Intensity: 3×10^{-5} Tesla for 200 to 1,000 km at Magnetic Equator
  6×10^{-5} Tesla for 200 to 1,000 km at Magnetic Poles
- Solar Intensity: 1,371 W/m^2 Earth Orbit
- Cloud Statistics: 30% of Measurements Randomly Compromised
- Ground Stations: NASA STDN S-Band Facilities (Longitude, Latitude)

Ascension Island (ACN)	345º 40' 22.57"	- 7º 57' 17.37"
Bermuda (BDA)	295º 20' 31.94"	32º 21' 05.00"
Guam (GWM)	144º 44' 12.53"	13º 18' 38.25"
Kauai (HAW)	200º 20' 05.43"	22º 07' 34.46"
Merritt Island (MIL)	279º 18' 23.85"	28º 30' 29.79"
Ponce de Leon (PDL)	279º 05' 13.12"	29º 03' 59.93"
Santiago (AGO)	289º 20' 01.08"	-33º 09' 03.58"
Wallops Island (WAP)	284º 31' 25.90"	37º 55' 24.71"

<u>Launch</u>

- Dimensions: To Be Determined
- Mass and Inertias: To Be Determined
- Resonant Frequencies: To Be Determined
- Vibration: SSTI Design Requirement (see Figure II-B.2 and Table II-B.1)
- Shock: SSTI Design Requirement (see Figure II-B.3 and Table II-B.2)
- Acoustic: SSTI Design Requirement (see Figure II-B.4 and Table II-B.3)
- Thermal: 10 to 24 deg C Prelaunch (Long Term), Max 125 deg C Ascent (Short Term),
  Max Rarefied Heating 400 BTU/hr ft^2 (SSTI DR)
- Pressurization: Sea Level Ambient to Vacuum at Rate of 0.35 psi/s (SSTI DR)

<u>Resources</u>

- Financial: $2 to 4 M (Estimated)
- Hardware/Software: To Be Determined

- Facilities: To Be Determined
- Testing: To Be Determined

<u>Technology</u>
- Attitude/Position Sensing: To Be Determined
- Moment/Force Generation: To Be Determined
- Impulse/Momentum Generation: To Be Determined
- Energy Conversion Efficiency: To Be Determined
- Energy Storage: To Be Determined
- Computational Capability: To Be Determined
- Communication Capability: To Be Determined

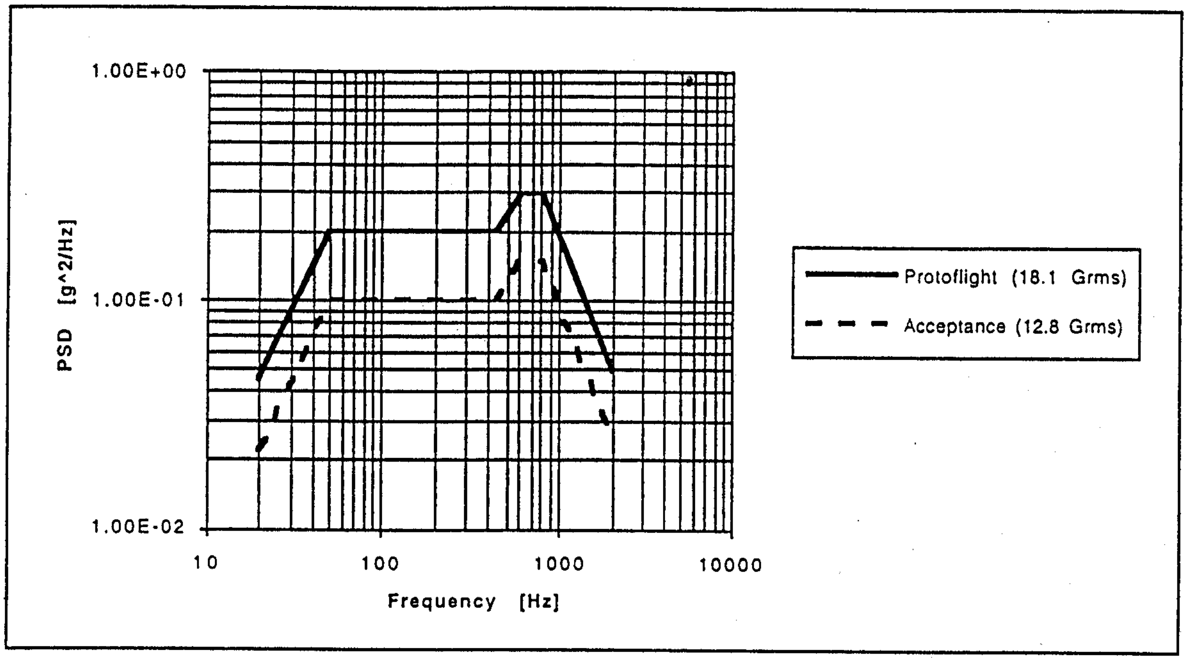

Figure II-B.2 SSTI Vibration Design Requirements

Table II-B.1 SSTI Vibration Design Requirements

Frequency (Hz)	Power Spectral Density (g^2/Hz)	
	Acceptance	Protoflight
20	0.0225	0.045
50	0.1000	0.200
440	0.1000	0.200
600	0.1500	0.300
800	0.1500	0.300
2,000	0.0250	0.050
RMS Average	12.8	18.1

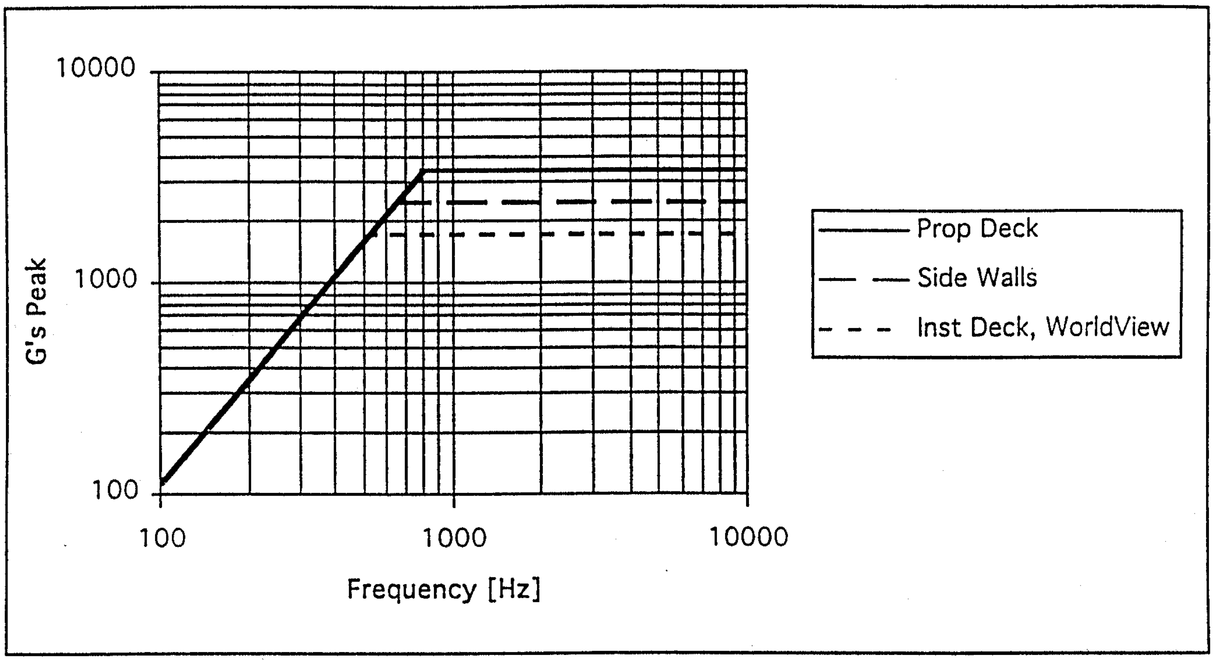

Figure II-B.3 SSTI Shock Design Requirements

Table II-B.2 SSTI Shock Design Requirements

Zone	Frequency (Hz)	Shock Response (g)
Prop Deck	100	110
	100 - 805	+ 5 db/oct
	805 - 10,000	3,470
Side Walls	100	110
	100 - 650	+ 5 db/oct
	650 - 10,000	2,430
Inst Deck / World View	100	110
	100 - 520	+ 5 db/oct
	520 - 10,000	1,700

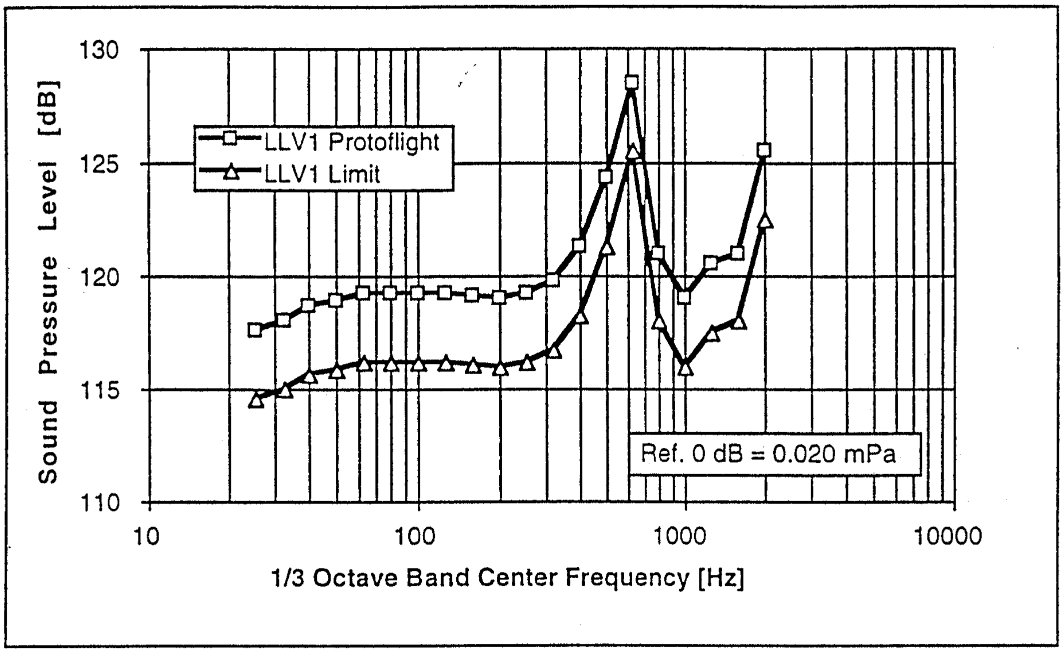

Figure II-B.4 SSTI Acoustic Design Requirements

Table II-B.3 SSTI Acoustic Design Requirements

1/3 Octave Center Frequency (Hz)	Noise Level (db) (Ref. Pressure = 0.02 mPa)	
	Acceptance	Protoflight
25	114.60	117.60
32	115.00	118.00
40	115.70	118.70
50	115.90	118.90
63	116.25	119.25
80	116.25	119.25
100	116.25	119.25
125	116.25	119.25
160	116.10	119.10
200	116.00	119.00
250	116.20	119.20
315	116.75	119.75
400	118.30	121.30
500	121.30	124.30
630	125.50	128.50
800	118.00	121.00
1,000	116.00	119.00
1,250	117.50	120.50
1,600	118.00	121.00
2,000	122.50	125.50
RMS Average	131.90	134.90

First consider development of requirement flow down relationships for Orbit Geometry. Only orbital altitude, inclination, synchronicity, and stability are considered in this analysis, and circular orbits are assumed exclusively. Figure II-B.5 shows the most significant mechanisms affecting requirements for these orbital geometry characteristics. Objectives and constraints from Science, Instrument, Environment, and Launch are the most significant factors here. Science objectives associated with high latitude coverage require orbit inclination angles above 75 deg. Environment constraints associated with drag from atmospheric density and complexities-expenses associated with shielding for Van Allen radiation require the orbit altitude to lie somewhere between approximately 200 to 1,000 km. No requirement seems to exist for temporal-spatial synchronous CO measurements. However, if one were imposed, a specific inclination-altitude interdependency would be required. Launch constraints for each opportunity will also impose requirements on orbital inclination and altitude, which are left unspecified at this time. To maximize Science data collection, the Instrument life rating imposes an additional mild requirement for orbital stability to maintain minimum acceptable altitude (200 km) and inclination (75 deg) conditions for at least 3 years. For a given orbit initialization, inherent natural stability will most likely be sufficient, but could be supplemented with a propulsion system. These flow down relationships are illustrated in Figure II-B.5. Resulting requirements are summarized below.

<u>Orbit Geometry</u>

- Inclination: Greater Than 75 deg
- Altitude: Greater Than 200 km, Less Than 1,000 km
- Synchronicity: None or Optional
- Stability: 200 km or Higher Altitude for 3 years
  75 deg or Higher Inclination for 3 years

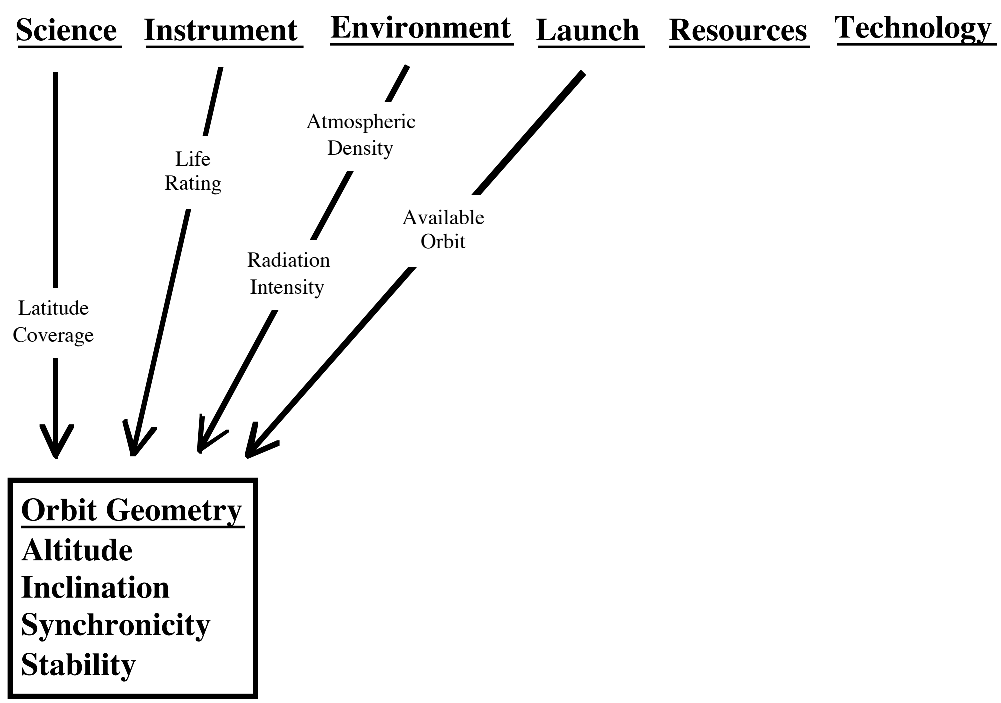

Figure II-B.5 Orbit Geometry Flow Down Relationships

Now consider development of requirement flow down relationships for Control. Within the control subsystem, only requirements for angular detection, moment generation, attitude authority, linear detection, and position authority are considered here. Force generation requirements are addressed under Propulsion. Figure II-B.6 shows the most significant mechanisms affecting requirements for these control system characteristics. Objectives and constraints from Science, Environment, Technology, Orbit Geometry, and several Subsystems (Telemetry and Other) are the most significant factors. Science objectives associated with CO measurement data fidelity and associated post-processing mandate knowledge of absolute instrument pointing and position to ±0.5 deg and ±25 km, respectively, while the accuracy of instrument pointing to a specified direction must be within ±5 deg. These objectives translate directly to requirements on angular detection, linear detection, and angular authority. These first two requirements (detection) impose conditions solely on the ability of sensor hardware to measure vehicle dynamic state information to sufficient precision (±0.5 deg and ±25 km). The latter requirement (authority) imposes a condition on the whole attitude control system (sensor, actuator, control logic, software, flight computer, etc.) to achieve and maintain a vehicle attitude state to within a specified tolerance (±5 deg). This requirement could impose further requirements such as a need for integral control logic to eliminate steady error in the presence of disturbances and sufficiently small nonlinear actuator traits like deadzones to prevent transients outside the ±5 deg limit. Note there is no direct requirement on position authority. However, orbit stability imposes a mild requirement for orbit inclination and altitude maintenance. Environment constraints associated with atmospheric density and gravitational disturbances influencing the spacecraft trajectory, as well as moment disturbances from atmospheric, gravitational and magnetic sources, require certain levels of force and moment generating capability from the control actuator hardware. Aerodynamic moment dominates below 400 km and requires a moment generation capability of 5×10^{-3} Nm at 200 km and decreasing to 8×10^{-5} Nm at 400 km, while magnetic moment dominates above 400 km requiring a constant 8×10^{-5} Nm moment level. These requirements are influenced by orbit altitude and inclination, as indicated in Figure II-B.6. Force generation requirements are considered under Propulsion. Based on the Telemetry data rate and storage requirements, and the frequency of downlink opportunities to ground stations which is influenced by Environment and Orbit Geometry factors (see Figure II-B.6), a requirement to periodically point to ground stations may be needed. Any related requirements for attitude maneuvers are left as "To Be Determined". Note inertias from the Other Subsystems (Structure) would strongly influence these requirements. Technology constraints impose additional requirements associated with the currently available capability vs. cost envelope, which are left unspecified at this time. All of these flow down relationships are illustrated in Figure II-B.6. Resulting requirements are summarized below.

<u>Control</u>

- Angular Detection: ± 0.5 deg
- Moment Generation: 5×10^{-3} Nm at 200 km (Aerodynamic)
  8×10^{-5} Nm at 400 km (Aerodynamic)
  8×10^{-5} Nm at 400 km and Above (Magnetic)
- Attitude Authority: ± 5 deg (Ref. 4 Lists ± 2.5 deg)
- Linear Detection: ± 25 km
- Force Generation: See Propulsion
- Position Authority: 200 km or Higher Altitude for 3 years
  75 deg or Higher Inclination for 3 years
- Attitude Maneuver: To Be Determined

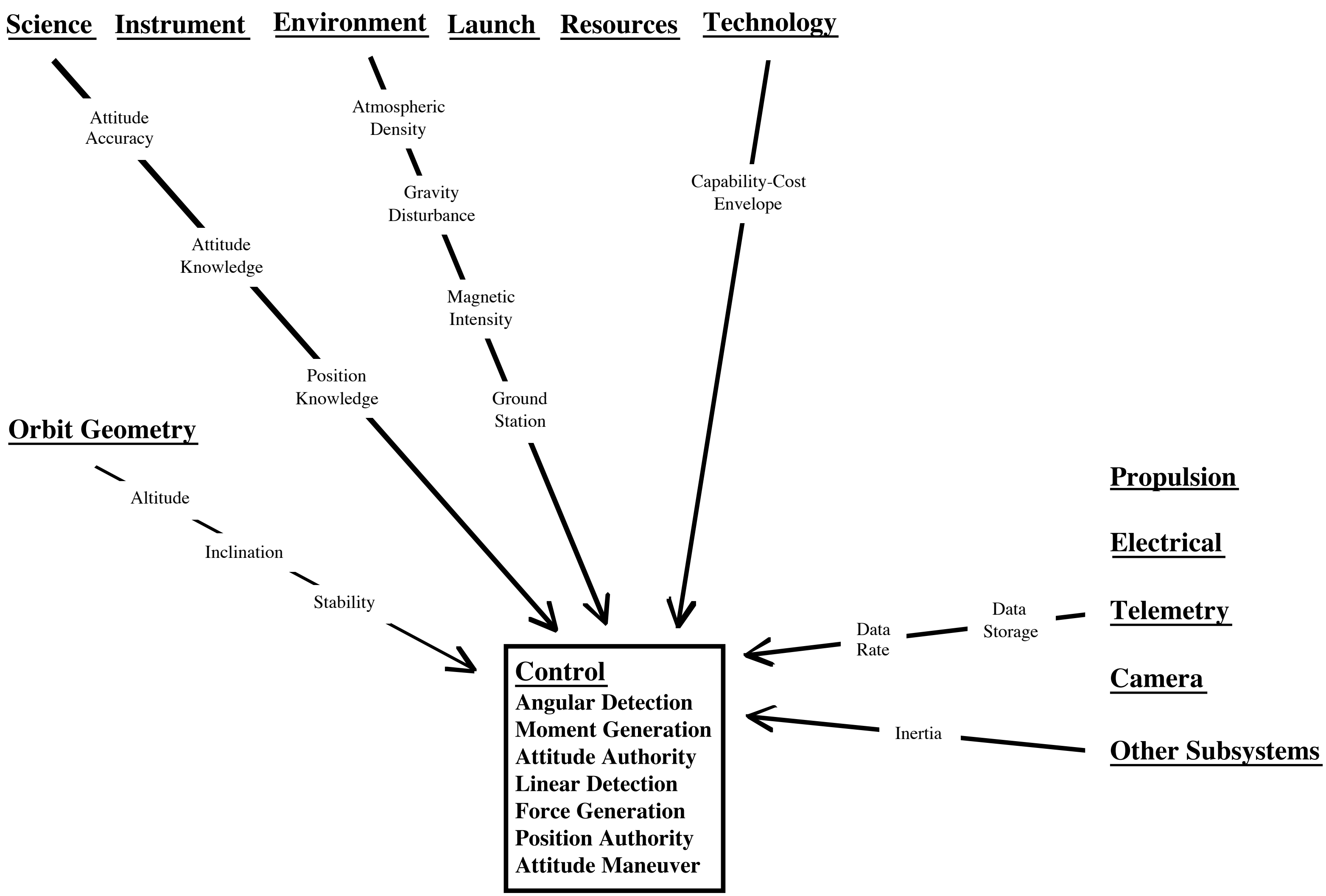

Figure II-B.6 Control Flow Down Relationships

Next consider development of requirement flow down relationships for Propulsion. Within the propulsion subsystem, only requirements for thrust level and total impulse are considered here. Figure II-B.7 shows the most significant mechanisms affecting requirements for these propulsion system characteristics. Objectives and constraints from Instrument, Environment, Technology, Orbit Geometry and Control are the most significant factors here. The primary function of the propulsion system is to maintain orbital altitude and inclination stability over the mission life. Inherent natural stability will most likely be sufficient for most orbit initializations lying within requirements noted previously. However, for initial orbit altitudes below approximately 300 km, depending upon the solar cycle phasing during the mission, the orbital decay rate compromises the mission before the 3 year instrument life is up. Orbital decay rate is computed by the method suggested in Reference 9 with an ample safety margin for uncertainty. Thus, a propulsion system is required for orbits below 300 km, and not required otherwise. A mission starting 3 to 5 years from the current time should experience a period of decreasing solar activity, lessening the need for a propulsion system. At the minimum acceptable orbit altitude of 200 km, the drag force is projected to be 0.021 N assuming the worst case atmospheric density, reference area of 1 m^2, and drag coefficient of 2. At 300 km the drag force would be 0.0015 N. Thus, Environment and Orbit Geometry constraints require a thrust level of at least 0.021 N at 200 km and 0.0015 N at 300 km, respectively, to maintain altitude. For a 3 year mission, these conditions translate to total impulse requirements of at least 1,987 kNs (200 km) and 141.9 kNs (300 km). These requirements are influenced by orbit altitude, stability, atmospheric density, solar activity, and position authority, as indicated in Figure II-B.7. Technology constraints impose additional requirements associated with the currently available capability vs. cost envelope, which are left unspecified at this time. All of these flow down relationships are illustrated in Figure II-B.7. Resulting requirements are summarized below.

<u>Propulsion</u>

- Thrust Level: 0.021 N for 200 km, 0.0015 N for 300 km, 0 N Above 300 km (Min)
- Impulse Total: 1,987 kNs for 200 km, 141.9 kNs for 300 km, 0 kNs Above 300 km (Min)

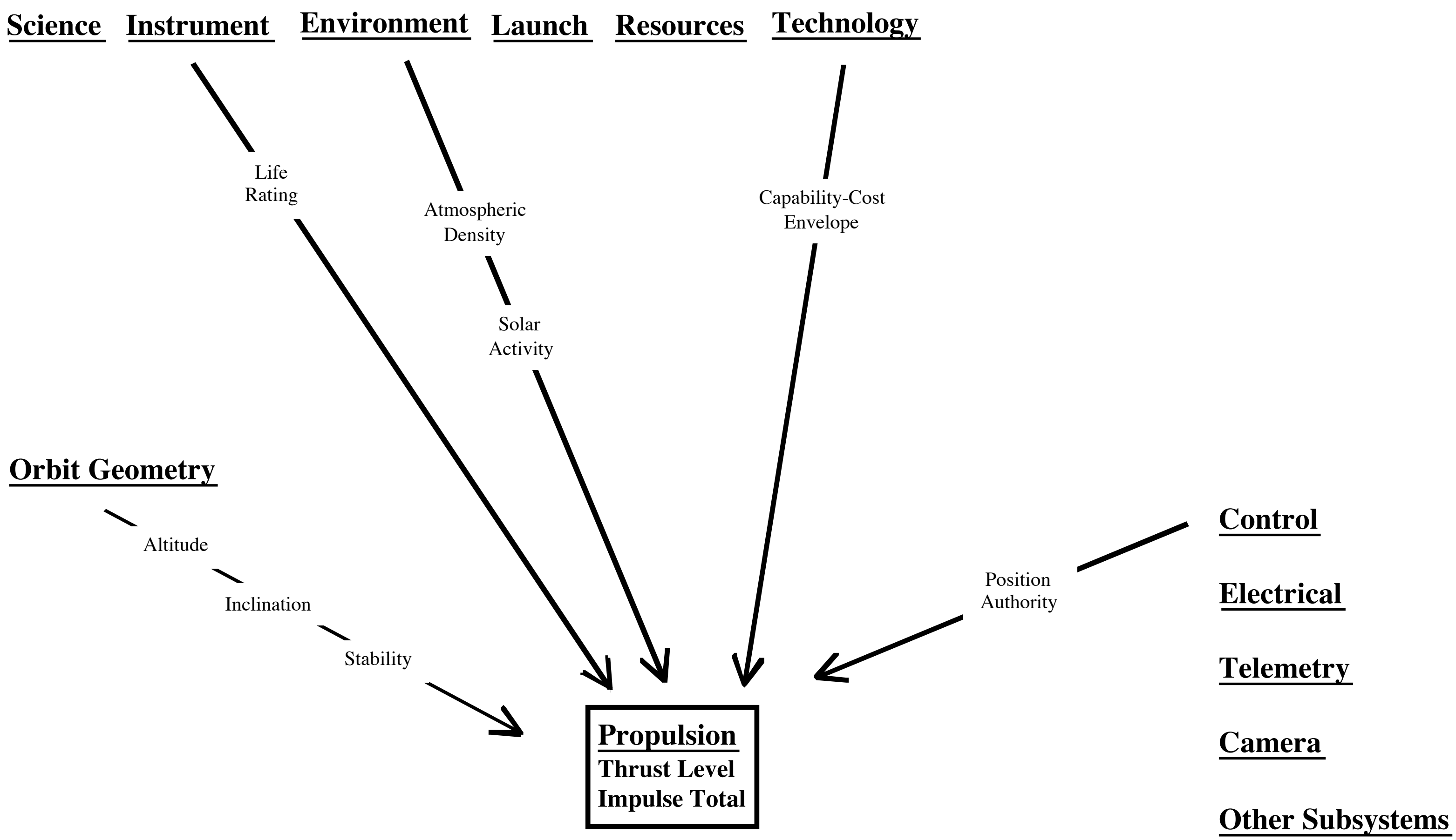

Figure II-B.7 Propulsion Flow Down Relationships

Next consider development of requirement flow down relationships for Electrical. Within the electrical subsystem, only requirements for power generation, energy storage, and surface area are considered here. Figure II-B.8 shows the most significant mechanisms affecting requirements for these electrical system characteristics. Objectives and constraints from Instrument, Environment, Technology, Orbit Geometry, and major power consumption Subsystems including Control, Propulsion, Telemetry, and Others (Thermal) are the most significant factors here. Power generation is one of the most straight forward requirements to be considered. An estimate of the system power budget translates directly to power generation demands. Total power consumption of approximately 300 W (no energy storage) is projected with contributions to the total consisting of 24 W for Instrument, 60 W for Control, 100 W for Propulsion, 10 W for Telemetry, and 100 W for Thermal. Therefore, a minimum requirement for 300 W power generation (assuming no energy storage) due to the Instrument and Subsystems is established, as indicated in Figure II-B.8. In Figure II-B.8, also note Orbit Geometry factors can influence the power generation requirement by determining the level of Control-Propulsion power consumption that is needed to maintain orbital altitude. There are two main options for generating this power: fuel cells or solar arrays. Fuel cell consumables and complexity may drive the spacecraft mass and design outside practical limits, and is therefore not considered further. Using spacecraft lighting estimates and solar energy conversion trends, requirement flow down relationships for energy storage and surface area can be further established. Spacecraft passage within the Earth shadow mandates a need for energy storage. Assuming an a-synchronous, high inclination low altitude orbit, the percentage of time corresponding to darkness is a worst case value of approximately 30%, or 0.45 hr for a 1.5 hr orbit period. Using a 10% nominal battery discharge depth, an energy storage requirement for 1,350 W hr is formulated. Note an additional 135 W of power generation capability is required, leading to a revised requirement of 435 W (including energy storage). Finally, assuming solar conversion efficiency of approximately 25% (a Technology constraint), a requirement for 1.27 m^2 of surface area is established. Resulting requirements are summarized below and Figure II-B.8 shows the flow down relationships.

Electrical

- Power Generation: At Least 435 W
- Energy Storage: At Least 1,350 W hr
- Surface Area: At Least 1.27 m^2

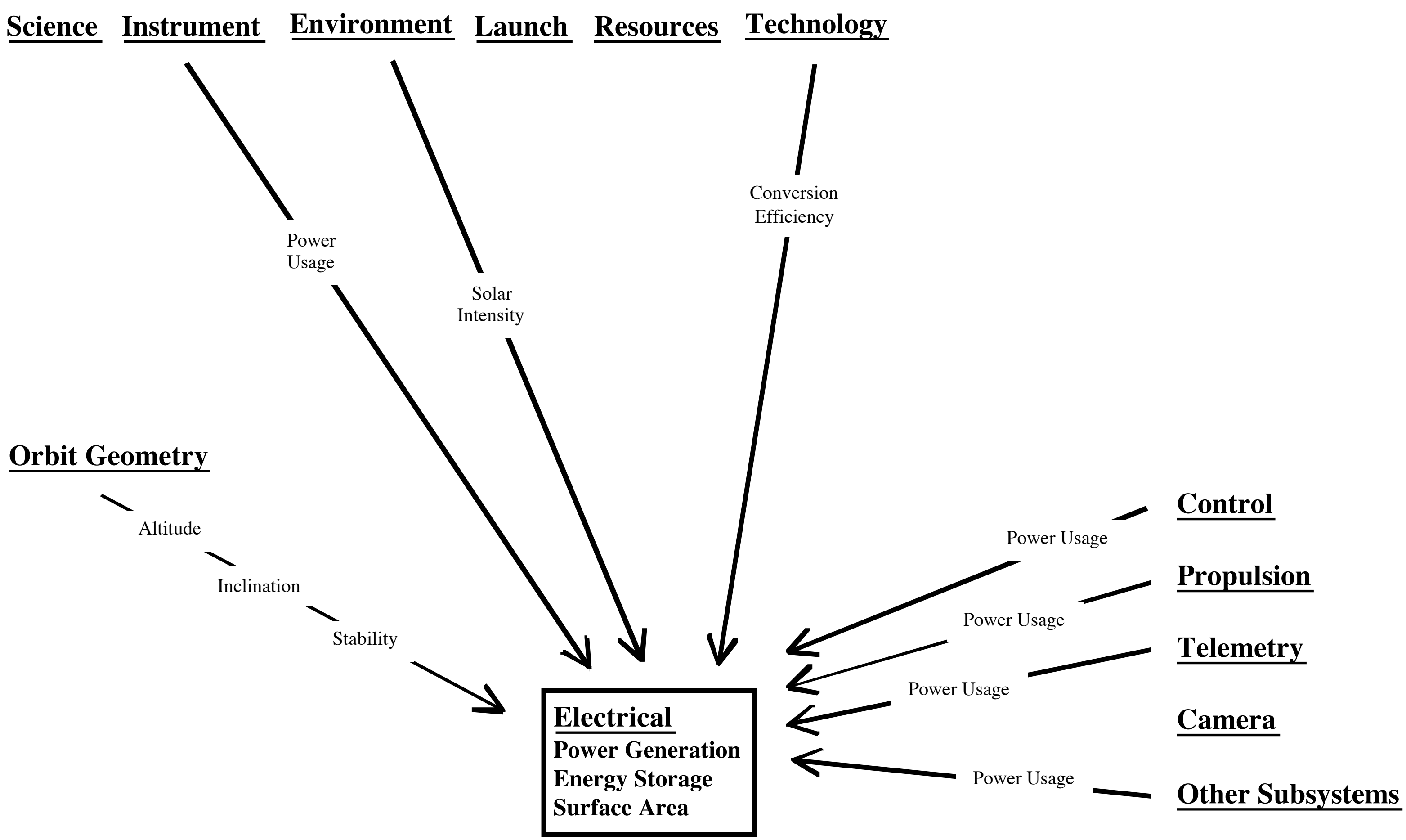

Figure II-B.8 Electrical Flow Down Relationships

Finally consider development of requirement flow down relationships for Telemetry. Only data rate, data storage, downlink power, and antenna gain are considered in this analysis. Figure II-B.9 shows the most significant mechanisms affecting requirements for these telemetry system characteristics. Objectives and constraints from Instrument, Environment, Technology, Orbit Geometry, and Camera are the most significant factors here. Instrument data generation rate after compression is 40 bit/s = 0.432 Mbyte/day. Further, the Instrument has a storage buffer capacity of 0.432 Mbyte. Thus, a minimum requirement for telemetry downlink data rate is 0.432 Mbyte/day (no camera). However, a maximum buffer content of only 25% at any given time is highly desirable to prevent scientific data loss if unexpected perturbations to the downlink were experienced. Thus, a more stringent requirement for data handling is 1.73 Mbyte/day (data rate) using the current storage buffer capacity. As discussed in Section III-E, if Earth image data of sufficient resolution must be downlinked also, the data rate and/or storage requirements could be much higher. Requirements for data handling with a camera are not considered here. Assuming a high inclination low altitude orbit with period of 1.5 hr, and based on the NASA STDN S-Band Ground Station geographic distribution and Earth spin rate, to ensure a downlink opportunity every 6 hr (0.25×24 hr) the downlink antenna beam width should be approximately 30 deg or larger. Assuming a conical beam shape, the corresponding antenna gain should be at least 60 = 35 db (see References 8-9). Using standard communication models for S-Band telemetry,[8,9] the product of antenna gain with transmitter downlink power is estimated to be 230 W. Thus, a minimum requirement for downlink power is 4 W. Higher data rate or lower antenna gain and downlink power requirements could be accommodated with attitude maneuvers for ground station pointing. Design freedoms of this type are not considered here. Figure II-B.9 illustrates these flow down relationships. Resulting requirements are summarized below.

<u>Telemetry</u>

- Data Rate: 1.73 Mbyte/day (no camera)
- Data Storage: 0.432 Mbyte
- Downlink Power: At Least 4 W
- Antenna Gain: At Least 60 = 35 db
- Uplink Sensitivity: To Be Determined

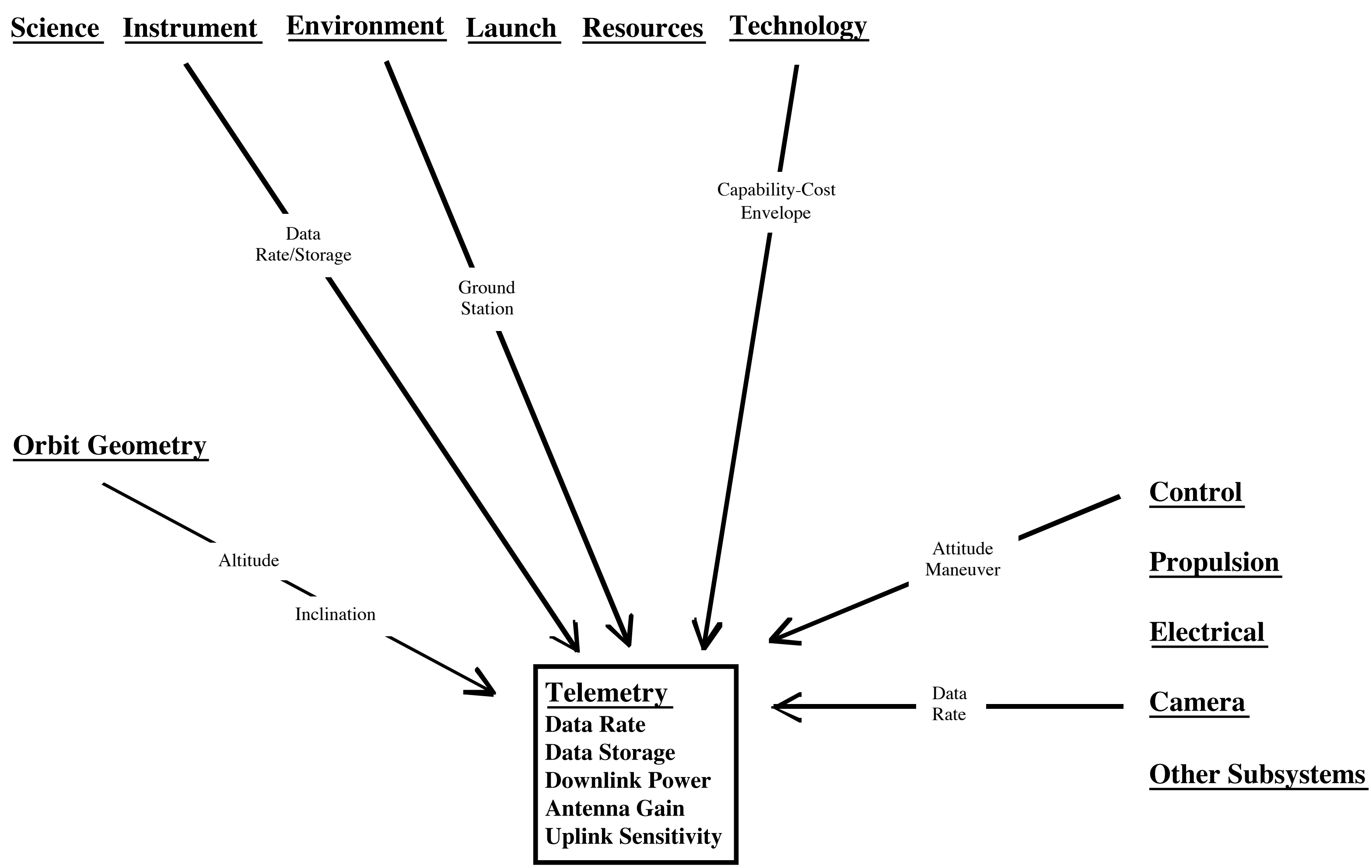

Figure II-B.9 Telemetry Flow Down Relationships

C. Orbit Selection

One of the basic mission analysis activities is to select the most suitable orbit for the mission. Mission orbit design usually proceeds in one of two approaches. The first approach is to calculate the orbit parameters based on the user requirements. Inputs to this approach are the payload and user requirements. For the MicroMaps Mission, these requirements are basically the spatial and temporal measurement resolutions. The second approach is to calculate the best achievable values for the user requirements based on a given available orbit. Inputs to this approach are the orbit parameters.

This subsection presents the first approach. Orbital parameters will be calculated based on the user requirements. An algorithm is developed to rapidly and roughly calculate a suitable orbit for a given set of requirements analytically. Only the J_2 gravitational perturbation is taken into account. A software tool that performs these calculations is built using Microsoft Excel spread sheets. Curves that illustrate the change in orbit altitude with variation of user requirements is presented. Two types of orbits are investigated. The first is the Earth-Sun synchronous orbit and the second is the Earth synchronous orbit. All orbits are assumed circular.

<u>Requirements</u>

Science objectives require the instrument to collect the CO distribution picture for the Earth at least once every season, or 90 days. However, more frequent CO distribution pictures for the Earth are certainly desirable. Define the period after which a new picture for CO distribution is obtained as the "Revisit Time". From the way by which the data of MicroMaps will be processed, one can deduce that no need exists to measure every point on the globe; rather the Earth surface is divided into boxes and the information for each box is considered uniform over the box. The size of a box is 5 deg longitude × 5 deg latitude. To get complete information about each box, the instrument needs to process at least 3 cloud free measurements in that box. The size of each box is equivalent to a rectangle with dimensions that will vary according to the latitude of the box. At the equator, the box dimensions, X_{LA} and X_{LO}, are approximately X_{LA} =

X_{LO} = 556.6 km. At latitude 80 deg, the rectangular dimensions are X_{LA} = 556.6 km and X_{LO} = 96.6 km (see Figure II-C.1).

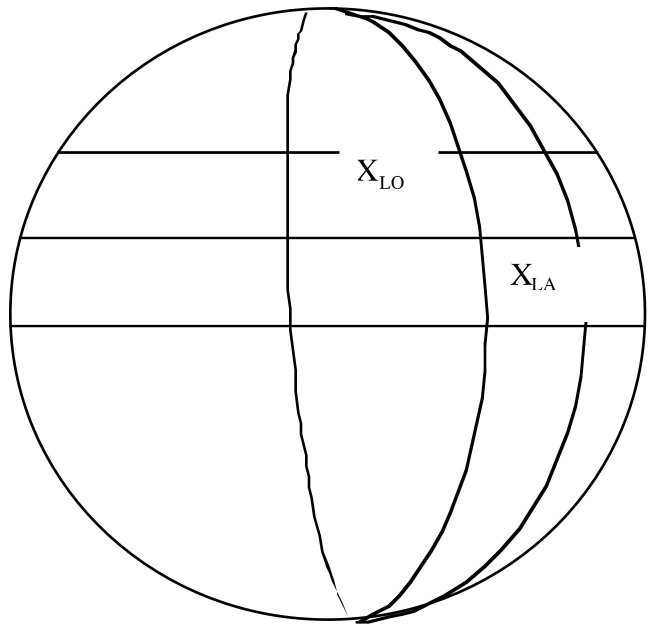

Figure II-C.1 Earth Surface Discretization

The number of data points required within each box is 3 cloud free measurements. A certain measurement cannot be expected to be cloud free or cloud obscured before it is measured, unless one uses statistical information, if available, to calculate the number of measurements, in each box, required such that at least 3 of them are cloud free. An approximate estimate for cloud statistics is that 30% of all measurements will be randomly obscured. Assume for the moment that 10 measurements per box are required so that at least 3 of them will be cloud free.

If it is sufficient to have a single path over each box in the revisit period, then the ground distance between tracks, i.e., the swath width, can be taken as 556 km at the equator. However, for more reliable performance, each box should be visited more than once in the revisit period. Assume that each box should be visited 4 times so that measurements can be obtained in any of the 4 visits. Thus, the swath width is around 120 km. Regarding the revisit time, a complete set of data will constitute a global picture for CO distribution and this set of data is likely to be obtained with at least a seasonal temporal resolution. Reasonable orbits can be found with revisit time periods of around 20 days.

Earth-Sun Synchronous Orbit

In this subsection a rough and rapid analytical approach is developed to get the orbit parameters that satisfies the required swath width and revisit time. Since the orbit is circular and Earth-Sun synchronous, defining the altitude will completely specify the orbit. The main idea is that an initial altitude is calculated based on a given swath width and revisit time taking into account only the condition of Sun synchronization. Then this initial altitude is corrected to the nearest altitude by applying the condition of Earth synchronization. A satellite flying at the new altitude will have a revisit time equal to that for the initial altitude but a slightly different swath width, as will be seen.

First, an initial altitude for the given swath width (Sw) and revisit time (m) are computed as follows. The distance on the ground between successive orbits (Dw) is related to Sw and m by

$$Dw = Sw \times m \tag{II-C.1}$$

The required change in longitude ΔL on the equator between successive orbits is

$$\Delta L = \frac{Dw}{R_e \cos(La)} \tag{II-C.2}$$

where R_e is the Earth radius and La is the latitude of the Earth location of interest. For a Sun synchronous orbit, Equation (II-C.2) can be expressed as

$$\Delta L = 2\pi \tau \left(\frac{1}{\tau_e} - \frac{1}{\tau_{es}}\right) \tag{II-C.3}$$

where τ is the satellite orbital period, τ_e is the Earth period through one revolution, and τ_{es} is the Earth orbital period around the Sun. For details on the preceding relationship derivations, refer to References 7-8. The required satellite orbit period τ can be calculated from Equation (II-C.3). τ is a function only of altitude, so the altitude (H) of the satellite can be computed from

$$\square = 2\square \sqrt{\frac{a^3}{\square}} \tag{II-C.4}$$

$$H = \sqrt[3]{\left(\frac{\square}{2\square}\right)^2 \square} - R_e \tag{II-C.5}$$

In Equations (II-C.4)-(II-C.5), $\square$ is the Earth gravitational constant and a is the orbit semi-major axis ($a = R_e + H$ for the assumptions made here).

Second, the condition of Earth synchronous orbit is checked to determine the appropriate altitude. This will be done as follow. It can be proved that for Earth-Sun synchronous orbits,

$$2\square n \sqrt{\frac{H^3}{\square}} \left(1 - \frac{\square_e}{\square_{es}}\right) = m \square_e \tag{II-C.6}$$

where n is the total number of orbits before an identical ground track occurs, m is the revisit time, and H is the altitude. Note variables m and n are integers. For the initial altitude, n is calculated. In general the calculated n will not be an integer which means that this altitude does not satisfy the condition of Earth-Sun synchronous orbit. So the nearest integer value for n will be taken to be the new proper value for n and calculate from Equation (II-C.6) the new altitude with the same value of m. In this way, a value for altitude that satisfies the condition of Earth-Sun synchronous orbit is obtained and is the nearest one to the requirements of the user. The new altitude is usually very near to the initially calculated one and resulting changes in user requirements are not significant.

Given the satellite altitude, the swath width is calculated as follows. The orbit period is calculated from Equation (II-C.4). The change in longitude on the equator is calculated from Equation (II-C.3). Finally, the swath width is calculated from combining Equations (II-C.1)-(II-C.2), or

$$Sw = R_e \square\square \cos(La) / m \tag{II-C.7}$$

Several numerical calculations are done using an Excel spreadsheet to calculate alternative altitudes for different values of swath width and revisit time. Figures II-C.2 and II-C.3 show some possible orbits for different user requirements.

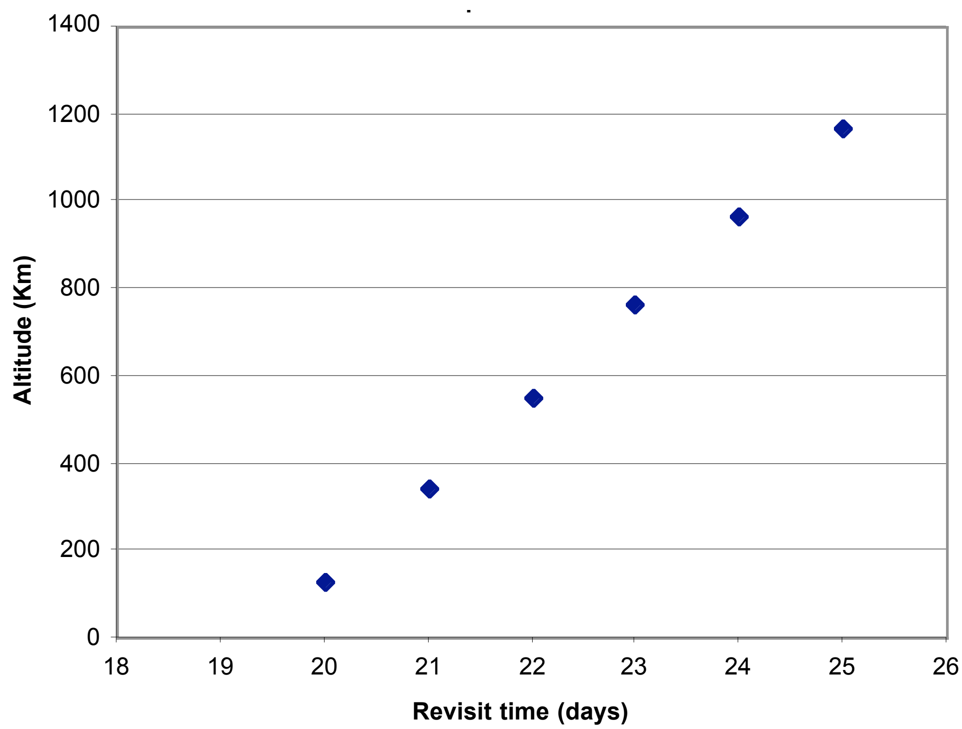

Figure II-C.2 Altitude vs. Revisit Time Chart (Swath Width = 121 km)

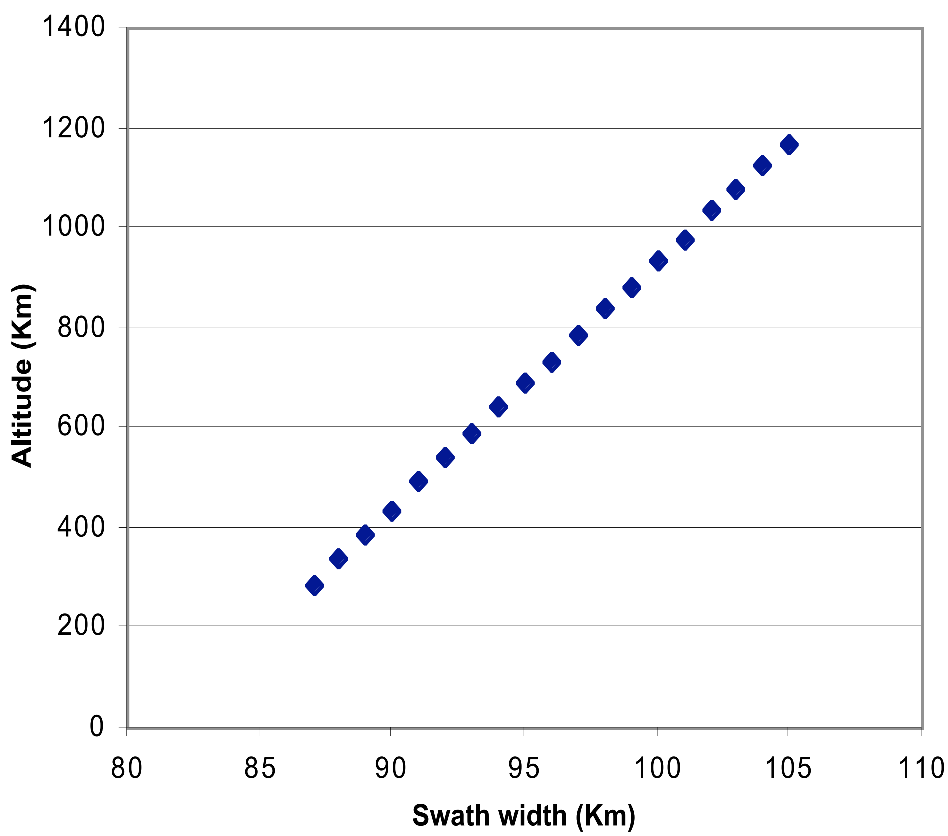

Figure II-C.3 Altitude vs. Swath Width Chart (Revisit Time = 25 days)

Ground Track Pattern

Results from the previous subsection showed that there are some orbits which are suitable for the MicroMaps Mission for the given requirements. In this subsection, the corresponding repass day pattern is determined. Repass day pattern means the number of days in which the satellite will pass over a certain area and the number of days in which the satellite will not pass over it. This information may be given in the following format, for example. For a certain orbit, the satellite will pass over a certain area in the first 2 days then it will not pass over it in the next 3 days, then it will pass over it in the next 2 days and so on. This information will be useful to select the most suitable orbit among the above possible orbits; since this information will determine the schedule by which the satellite will pass over certain ground stations or any ground object. Repass day pattern is a criterion to select among the possible orbits. The next analysis determines the basic concept of how this criteria will be calculated.

A typical ground track is plotted in the Figure II-C.4. Assume that the satellite passes over track 1 and track 18 in the same day. The satellite passes over the tracks 2, 3, 4, 5…17 in the following days. If the satellite passes over track 2 in the second day and on track 3 on the third day and so on, the orbit of the satellite is called a minimum drift orbit. If the satellite passes over

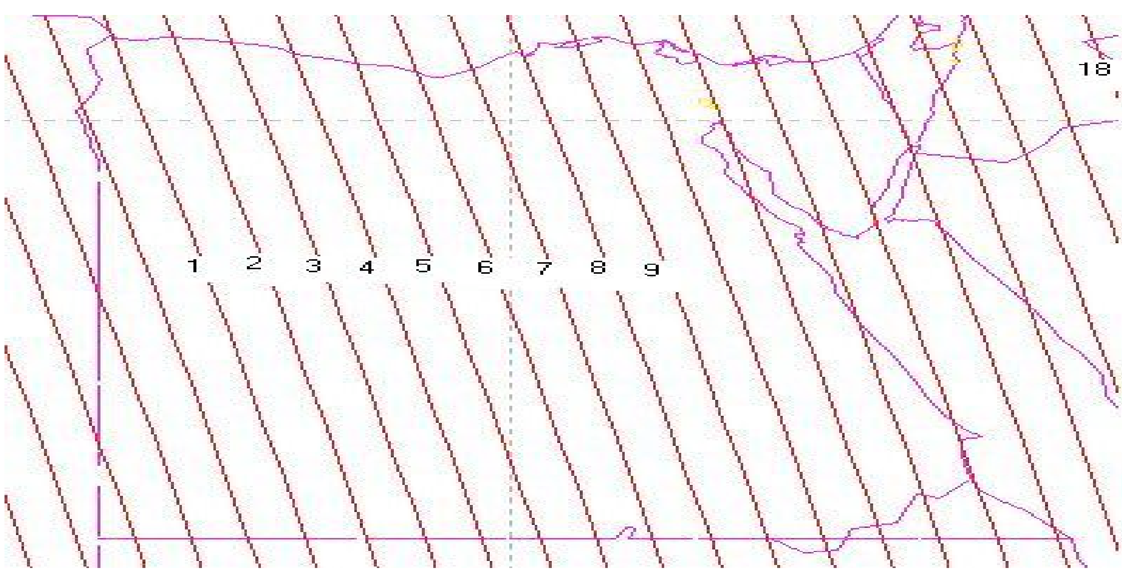

Figure II-C.4 Typical Ground Track for Low Earth Orbit Satellite track 2 in the second day and on track 5 in the third day or in any other order of tracks in the subsequent days to the first day, the orbit of the satellite is called a non-minimum drift orbit. For a minimum drift orbit, the repass day pattern is obvious. If for example, the whole period of revisit time is 53 days, the satellite will pass over a certain area every day for certain number of days and then does not pass over it for the rest of the period of revisit time. For a non-minimum drift orbit, some calculations must be done to determine the repass day pattern. These calculations are considered next.

Let the number of orbits that a satellite performs in one day be n. In general, n is not integer. Since the satellite is orbiting in an Earth-Sun synchronous orbit, then the satellite will revisit a certain point on the ground every certain number of days, let it be M days. M is integer. During these M days, the satellite will perform N orbits. The condition of Sun synchronization implies that N is an integer also.

$$n = \frac{N}{M} \tag{II-C.8}$$

Now, assume that $n = j + i$ where j is an integer which represents the number of complete orbits performed in one day. Parameter i is a fraction less than 1, let it be K/M. This parameter represents the part of the orbit, which is performed after j orbits are performed to complete one day of orbiting. As an example, if N = 800, M = 53, then n = 800/53 = 15 + 5/53. Thus, j = 15 and i = 5/53. After a complete day of orbiting, the satellite performs a complete 15 orbits plus 5/53 of an additional orbit.

Now, return to Figure II-C.4. The satellite will pass on track 1 and on track 18 in the same day; it will pass on track 1 in the first orbit and on track 18 in the second orbit of the same day. The distance on the ground between track 1 and track 18, call it S, is then the distance scanned in one orbit of the satellite motion. After one day the satellite will not pass on track 1 but on a track which is shifted from track 1. This shift is due to the fraction i of the orbit, which a satellite performs to complete one day of orbiting. If i = 0, the satellite will repeat track 1 after one day.

Thus, after one day, the satellite will pass on a track which is shifted a distance i □ S on the ground from track 1. After two days the satellite will pass on a track which is shifted a distance 2i □ S from track 1. After M days the satellite will pass on a track which is shifted a distance Mi □ S from track 1. Recall that Mi = K, which is an integer value.

Now, assume (without loss of generality) that the first track of the first day passes over the area under consideration. One can calculate the pattern of repass days as follow. Calculate the distance shift of the first track in the second day from the first track in the first day and check if it is within the band of that area or not, and repeat for the first track of the third, fourth, … day till the satellite completes the period of revisit time M. These calculations are programmed on an Excel spreadsheet. As an example, a certain area of 1,000 km □ 1,000 km is considered and the results are plotted in Figure II-C.5. In Figure II-C.5, N is the maximum number of days of not visiting the area, V is the total number of days of visiting the area, T is the total number of days of not visiting the area, and T + V is the revisit period.

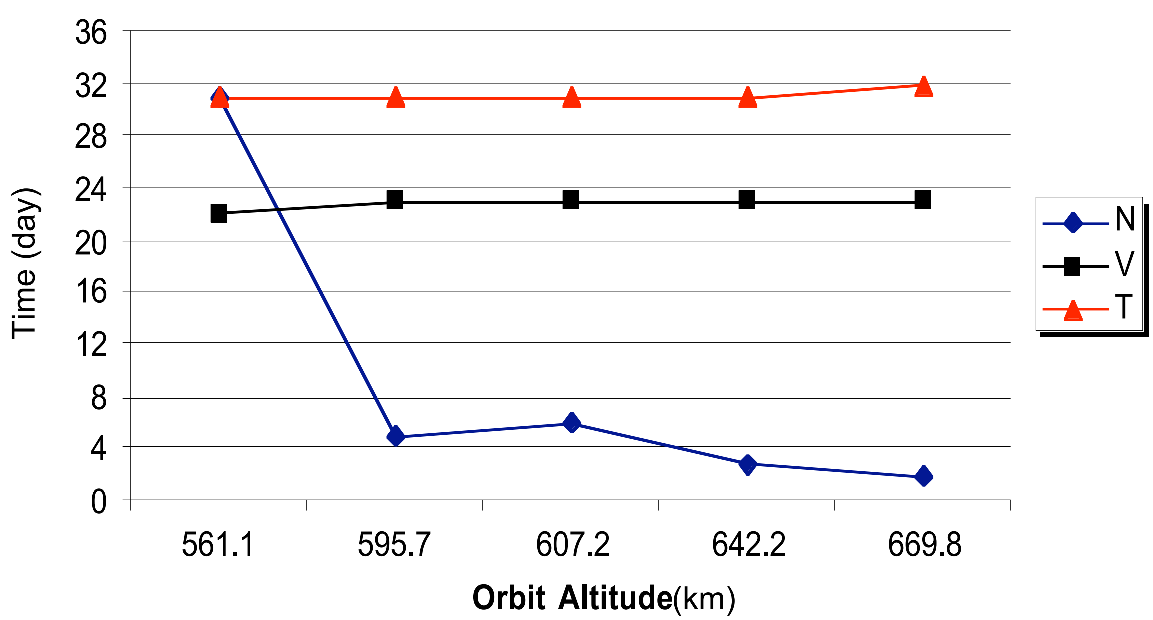

Figure II-C.5 Ground Track Pattern for Various Altitudes

Earth Synchronous Orbit

For the MicroMaps Mission, it is not scientifically required to fly the instrument in a Earth-Sun synchronized or Earth synchronized orbit. However it could be advantageous to fly the instrument in an Earth synchronous orbit for engineering purposes. In this case, the number of equations is less than the number of unknowns (for circular orbits) yielding many solutions for a single set of user requirements. This fact is especially important considering the launch conditions are not well defined at this time. In this subsection, a quick and rough approach is developed to calculate the possible orbits for a single set of requirements. This tool is developed using an Excel spreadsheet.

The mathematical algorithm starts by specifying the requirement set for revisit time and swath width. The initial steps are to calculate Dw from Equation (II-C.1) and $\Delta\lambda$ from Equation (II-C.2). Next compute n using Equation (II-C.9).

$$n\,\Delta\lambda = 2\pi m \tag{II-C.9}$$

Next correct n to the nearest integer. Recompute $\Delta\lambda$ using Equation (II-C.9), recalculate Dw with Equation (II-C.2), and recompute Sw using Equation (II-C.1). Now select a value for orbit altitude H, and compute the orbit inclination i for the specified H using the following relationships.

$$\tau = 2\pi\sqrt{\frac{a^3}{\mu}} \tag{II-C.10}$$

$$\Delta\lambda_1 = 2\pi\,\tau\left(\frac{1}{\tau_e}\right) \tag{II-C.11}$$

$$\Delta\lambda_2 = \Delta\lambda - \Delta\lambda_1 \tag{II-C.12}$$

$$\dot{\Omega} = \Delta\lambda_2 / \tau \tag{II-C.13}$$

$$\cos(i) = -\dot{\Omega}\,\frac{2}{3}\,\frac{a^2}{2\pi/\tau}\,\frac{(1-e^2)}{R_e^2\,J_2} \tag{II-C.14}$$

For circular orbits, eccentricity e will equal zero. A family of solutions is obtained by using different values for H. This algorithm is implemented on an Excel spreadsheet and Figure II-C.6 illustrates results for selected cases.

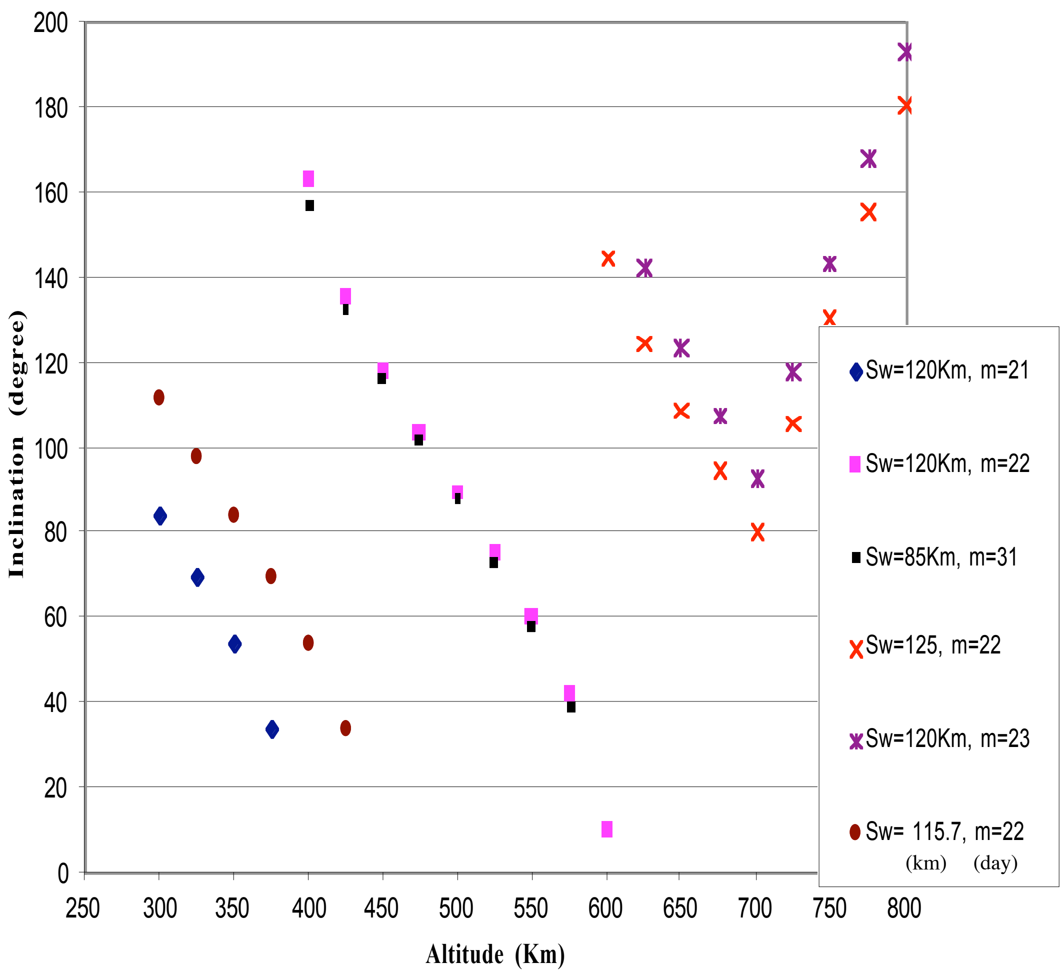

Figure II-C.6 Candidate Earth Synchronous Orbits for MicroMaps

Section III
Dedicated Spacecraft - Subsystem Studies

A. Attitude Sensing and Control

The main task of the Attitude Determination and Control System (ADCS) is to counter-act the disturbance torques that affect the satellite in its space environment. It should also provide the required torque to conduct necessary maneuvers within a mission. In this subsection, the ADCS of the satellite whose primary mission is to carry MicroMaps will be investigated. Recommendations will be given at the end of the subsection. Two mission options will be considered. The first will be sending MicroMaps on a dedicated satellite to orbit. The moments of inertia of this satellite are assumed to be 20, 15 and 10 kg m^2 in the x, y, and z directions. The second option will be sending a satellite whose primary mission is MicroMaps, and with a secondary mission of carrying a camera. The moments of inertia of the second satellite are assumed to be 30, 25 and 20 kg m^2 in the x, y, and z directions. The increase in inertia accounts for additional hardware such as the camera and antennas, and a larger power generation system to operate these hardware components. A range of orbits from 200 to 1,000 km will be considered. This range is considered to facilitate finding a suitable orbit.

Disturbance torques are either due to internal sources, such as due to misalignment of thrusters or sloshing in fuel tanks, or due to external sources. There are four main sources of external torques; solar pressure, gravity gradient, Earth magnetic, and atmospheric density. These external torque sources will be discussed next.

Solar Pressure Torque

This torque arises from the difference in the force produced by the impact of the solar rays on various surfaces of the satellite. It is dependant on the type of surface, on its area and on the distance between the center of mass and the center of pressure of this surface. A symmetric satellite will probably be affected by a negligible solar torque. In general, the torque is given by

$$T_s = \frac{F_s}{c} A_s(1+q)\cos(i)\,(C_s - C_m) \qquad \text{(III-A.1)}$$

where

- F_s Solar Constant (1,371 W/m^2)
- c Speed of Light (3×10^8 m/s)
- A_s Surface Area. (1 m^2, Conservative Estimate)
- q Reflectance Coefficient (0 to 1 Range, 0.6 Typical Value)
- i Angle of Incidence for Sun Vector (0 deg for Max Torque)
- C_s Position of Solar Pressure Center (Dependent on Spacecraft External Shape)
- C_m Position of Mass Center (Dependent on Spacecraft Mass Distribution)

The quantity C_s - C_m is the moment arm and will be assigned a value of 0.2 m. This parameter could be potentially very high, if a hexagonal satellite configuration is chosen. It is clear that the solar pressure torque is not affected by altitude, but rather by the geometric configuration of the satellite. This makes it very difficult to calculate torque accurately unless a complete design of the satellite is available. Consistent with the preliminary investigation stage of this report, and using the above assumed numbers, it was found that the solar pressure torque is on the order of 10^{-6} Nm, at its maximum.

<u>Gravity Gradient Torque</u>

This torque arises from the gradient in the gravitational attraction force of the Earth along the length of the satellite in the direction of the Earth. This gradient is very small, but with the practically frictionless environment in space, it introduces a disturbing torque that must be accounted for. The larger the satellite the larger this torque is. In general, the torque is given by

$$T_g = \frac{3}{2}\frac{\mu}{R^3}(I_{zz} - I_{yy})\sin(2\theta) \qquad \text{(III-A.2)}$$

where

- I_{yy} Moment of Inertia about y Axis (15 to 25 kg m^2)
- I_{zz} Moment of Inertia about z Axis (10 to 20 kg m^2)

R   Orbit Radius (200 to 1,000 km Plus Earth Radius)
θ Angle Away From Nadir (5 deg for Max Torque)

It is found that the gravity gradient torque is of the order of 10^{-6} Nm. This torque was calculated for both of the satellite configurations considered. The higher the orbit, the less significant this torque.

Earth Magnetic Torque

The electric wiring in a satellite produces an internal electric field, which interacts with the Earth's magnetic field to produce a torque. In general, this torque is given by

$$T_m = DB = \begin{cases} D\,2\dfrac{M}{R^3} \text{ (At Magnetic Poles)} \\ D\,\dfrac{M}{R^3} \text{ (At Magnetic Equator)} \end{cases} \tag{III-A.3}$$

where

B   Earth Magnetic Field Intensity
D   Spacecraft Magnetic Dipole Moment (1 Am^2, Standard for Small Satellites)
M   Earth Magnetic Moment (7.96×10^{15} Tesla m^3)
R   Orbit Radius (200 to 1,000 km Plus Earth Radius)

The magnetic disturbance torque is highly dependant on the altitude. Torque order of magnitude is 10^{-5} Nm in orbits ranging from 200 to 1,000 km, one order of magnitude higher than any of the other disturbance torques (except for very low altitude aerodynamic torque). The magnetic torque was calculated based on polar (or Sun synchronous) orbits, i.e., based on the worst case. If the launch opportunity yields an inclination of 60 or 70 deg, then the value of this disturbance would be half as much as its value in the current case. Even with a higher disturbance, a polar orbit is preferred to a less inclined orbit due to mission requirements.

Atmospheric Density Torque

The aerodynamic drag from the upper atmosphere density on the uneven surfaces of the satellite, causes a disturbance torque that tends to change the attitude of the satellite. In general, this torque is given by

$$T_a = \frac{1}{2} \rho V^2 A C_D (C_a - C_m) \tag{III-A.4}$$

where

ρ	Atmospheric Density (2.54×10^{-10} to 3.561×10^{-15} kg/m^3)
V	Orbital Velocity (Circular Orbit Assumed)
A	Projected Area (1 m^2, Large Value for Compact Spacecraft)
C_D	Drag Coefficient (2.5 Worst Case)
C_a	Position of Aerodynamic Pressure Center (Dependent on Spacecraft External Shape)
C_m	Position of Mass Center (Dependent on Spacecraft Mass Distribution)

The quantity C_s - C_m is the moment arm and will be assigned a value of 0.2 m, which may be an over estimation. The aerodynamic torque is in the order of 10^{-6} Nm at 500 km and decreases to the order of 10^{-8} Nm at 800 km. These values were calculated using the maximum density of the atmosphere at the corresponding altitude. If solar panels are not used, and if the center of pressure is made as close as possible to the center of mass, these values can be lowered even more.

Torque Comparison

All the disturbance torques versus altitude are plotted in Figure III-A.1. It is very clear that the magnetic torque values are to be used in the design process. All other torques, even if summed together, are negligible compared to the magnetic torque. This observation is true for orbits higher than 400 km. For lower orbits (highly unlikely), the aerodynamic torque is the driving factor. Because the magnetic torque is relatively invariant to altitude (see Figure III-A.1), and does not depend strongly on satellite configuration, altitude is not a significant driving factor in choosing an orbit for this mission based on ADCS disturbance rejection considerations. Due to the nature of the mission, a limited number of orbits might be available. The disturbance torque is not a limiting factor in the choice of the orbit. This observation must not be confused with the fact the mission is already limited to available launches within a 3 to 5 year window referenced to the current time heading towards pre-specified orbits.

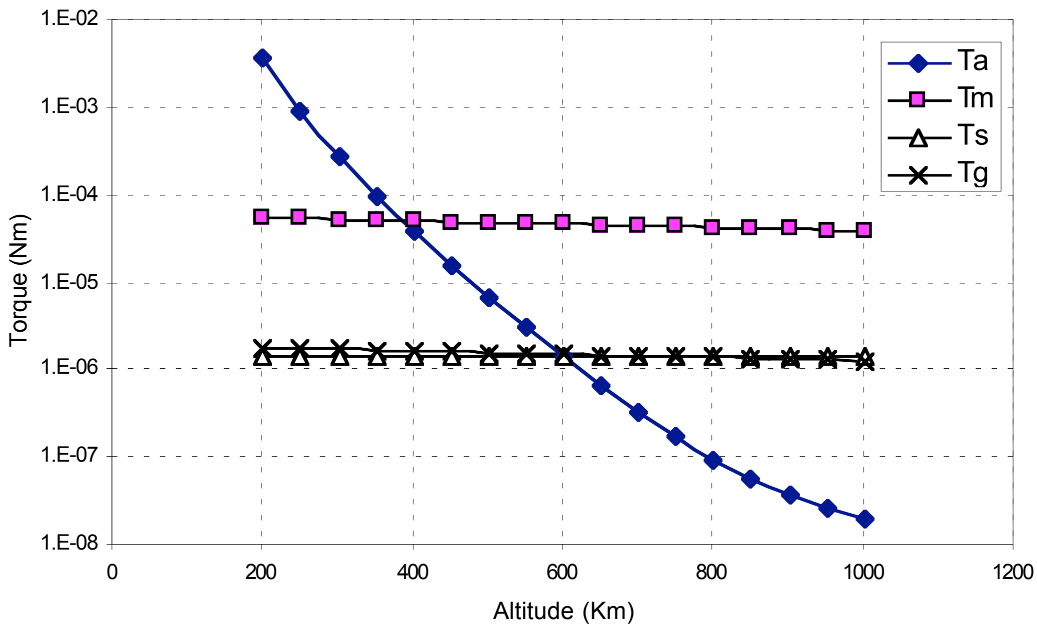

Figure III-A.1 Disturbance Torques Affecting Satellite Pointing

Hardware Selection

The choice of the hardware depends on both the mission requirements and inertia properties of the satellite. A suitable hardware configuration for a small 3-axis stabilized satellite uses three reaction wheels for attitude changing (a fourth redundant reaction wheel is also added), three torque rods or magnetic coils for momentum dumping from the reaction wheels, a 3-axis magnetometer, and a pair of attitude sensors such as a Sun sensor and an Earth sensor (another pair is also used for redundancy). If the rates are to be measured (rather than calculated using differentiation algorithms from position sensors), then rate gyros must also be added. An alternate configuration uses the torque rods for attitude control and discarding the reaction wheels. This arrangement is a suitable configuration only for very small satellites in low Earth orbits. An algorithm must be developed for the attitude control system. This system might be adequate to the first mission option, i.e., the option that builds a satellite dedicated entirely for MicroMaps without a camera.

For a reaction wheel to be chosen, one must determine the maximum torque to be delivered by the reaction wheel to counteract a disturbance, and the maximum amount of momentum it will store for a quarter orbit for cyclic disturbances. The angular momentum capacity on a reaction wheel is given by Equation (III-A.5) where H denotes angular momentum and τ denotes orbital period.

$$H = T_d \frac{\tau}{4} (0.637) \tag{III-A.5}$$

Applying this relationship to our first mission option, and assuming a disturbance of 5×10^{-5} Nm (which comes from the magnetic torque estimation), it follows that a reaction wheel must deliver a torque of this same value for proper control. This level is not a large torque and can be achieved by a small reaction wheel. The angular momentum capacity is dependant on the orbital period, hence the altitude. Orbits ranging from 400 to 800 km have orbital periods between 92.6 and 101 min. This range gives reaction wheels of a maximum of 0.05 Nms. This level is a very small value, and most of the commercially available reaction wheels will satisfy this requirement with a very generous operational margin.

In the second mission option, the satellite will be required to conduct tilt maneuvers in a given time for observation or downlink purposes. The overall angular motion per required time is known as the slew rate. The torque to conduct a constant slew rate maneuver is given by

$$T = 4\theta \frac{I}{t^2} \qquad \text{(III-A.6)}$$

This torque also depends on the inertia I around the slew axis (the x axis in our case). If it is assumed that the satellite is required to tilt at a rate of 1 deg in 10 seconds (which is reasonable), the required torque will be 0.02 Nm. This level is a large torque compared to disturbance torque, but can be satisfied easily with commercially available momentum wheels, also with an ample operating margin.

Either torque rods or magnetic coils can be used to dump the momentum build up on the reaction wheels. These devices can also be used as the primary actuator for attitude control. If they are used in this way, they must be able to counteract disturbance torques. The size of the torque rod is given by

$$D = \frac{T}{B} \qquad \text{(III-A.7)}$$

where B is the worst case available magnetic field (6×10^{-5} Tesla at the poles and 800 km). This value will yield a 1.3 Am^2 torque rod (for $T = 8 \times 10^{-5}$ Nm). This value is less than realistic, since the magnetic field changes both in magnitude and direction, and might not always be available for usage in attitude control. A more appropriate value would be 5 Am^2. Yet, if the only function of the torque rods is to dump the momentum wheels, this value would be far more than enough.

It should be noted at this point that lower inclination orbits have less magnetic field available around them, which makes it a hard task to design a satellite using torque rods solely for attitude control. The sensor selection depends on the accuracy required. For MicroMaps, this is not a stringent requirement. Two Sun sensors and two Earth sensors can be used for both mission options, yielding an accuracy of about 1 deg in low Earth orbit. This accuracy is sufficient for satisfying the camera and MicroMaps requirements. It is also to be noted that for the first mission option, i.e., for MicroMaps alone, a simpler hardware configuration can be employed. This arrangement consists of a momentum wheel, and three torque rods. The accuracy in pointing the satellite would still remain at 1 deg if a suitable momentum wheel is chosen.

In summary, two mission options are considered. Both are feasible, but very different in hardware selection. A decision should be made regarding whether MicroMaps is to have a dedicated mission, or is to have a primary mission in addition to some other secondary mission. The ADCS of the first mission option is more relaxed than that of the second option. Yet, the second mission option can provide increased probability for funding by enhancing the educational component. The orbit altitude proved to be a key point in determining the disturbances affecting the satellite (magnetic or atmospheric). Altitude should be determined as early as possible in the design process. The above study was conducted over a family of orbits ranging from 200 to 1,000 km. The orbit determining factor will actually be the launch opportunities available for the mission. Orbits with higher altitudes will be more expensive regarding launch, they will introduce better coverage, and will clear the satellite from aerodynamic torque and drag. Lower orbits will be cheaper launch-wise, but will subject the satellite to aerodynamic drag and moment, making the orbit decay faster. Orbits with higher inclinations are preferred. A polar or Sun synchronous orbit is as good as it gets. Full Earth coverage is achieved, and the opportunity to use the stronger polar magnetic field in attitude control is available. Yet, polar orbit launches cost more and are not accessible from many launch sites, and not accessible using the Space Shuttle. Several hardware configurations were suggested. The first is an all reaction wheel system. The second uses an all torque rod system. The third uses a momentum wheel and torque rods along the other two axes. An estimate of power consumption of the ADCS is projected to be around 60 W. No rate gyros were considered in this estimate. These conclusions are based on assumed data of the moments of inertia. In most of the scenarios, worst case operating conditions are assumed. Iterations will be required to reach a final design for the ADCS.

B. Orbital Adjustment and Maintenance Propulsion

Satellites in low Earth orbits experience enough atmospheric drag to decay their orbit within a matter of years, and increase difficulty in ground-based tracking of the vehicle. Also, this decay can affect data readings for Earth observing satellites, since a lower orbit will cause the satellite to travel over the Earth quicker giving less time for ground observations. Additionally, gravitational perturbations from the Earth will cause the orbit to slowly drift over time away from the desired geometry. In order to extend the useful life of a satellite and avoid the problem of shifting orbits, a propulsion system can be added to the satellite as a way of countering drag and gravitational perturbations. Before analyzing propulsion system characteristics, the amount of drag or impulse loss due to the atmosphere in low Earth orbit is considered.

Spacecraft in low Earth orbit experience low levels of atmospheric drag due to the presence of a tentative atmosphere. The equation for atmospheric drag, D, is

$$D = \frac{1}{2}\rho V^2 S C_D \tag{III-B.1}$$

where C_D is the drag coefficient which ranges from 2.8 to 4 depending on the shape of the spacecraft, ρ is the atmospheric density, V is the velocity of the spacecraft in orbit, and S is the cross-section area of the vehicle normal to the velocity vector of the spacecraft. Average atmospheric density ranges from 2.5×10^{-10} to 1.9×10^{-13} kg/m^3 (200 to 700 km). The atmospheric density varies with time depending on the time of day and the time of the solar cycle, the 11 year cycle of Sun spot activity. Density increases during the day and during the height of the solar cycle. Since the solar cycle reaches its peak in 2002, solar activity should decrease for the next five to seven years, allowing MicroMaps to enjoy a period of lower atmospheric drag. Tables III-B.1-III-B.4 are charts showing the yearly and daily changes in the density at altitudes of 400, 500, 600, and 700 km, respectively, and Figure III-B.1 is a graph of the annual change in density over time from a period of 1959 to 1962, where 1958 was the peak of the solar cycle.[11] Data for 200 and 300 km altitudes can be found in Section II-B.

Table III-B.1 Atmospheric Density at 400 km Altitude

Year	Density (kg/m^3)		
	Day	Night	Average
1959 (0)	9.00×10^{-12}	5.00×10^{-12}	7.93×10^{-12}
1960 (1)	7.00×10^{-12}	3.00×10^{-12}	5.91×10^{-12}
1961 (2)	4.10×10^{-12}	1.30×10^{-12}	3.33×10^{-12}
1962 (3)	3.00×10^{-12}	7.00×10^{-13}	2.36×10^{-12}

Table III-B.2 Atmospheric Density at 500 km Altitude

Year	Density (kg/m^3)		
	Day	Night	Average
1959 (0)	3.00×10^{-12}	9.50×10^{-13}	2.44×10^{-12}
1960 (1)	2.00×10^{-12}	4.20×10^{-13}	1.56×10^{-12}
1961 (2)	7.60×10^{-13}	2.00×10^{-13}	6.05×10^{-13}
1962 (3)	4.20×10^{-13}	8.00×10^{-14}	3.26×10^{-13}

Table III-B.3 Atmospheric Density at 600 km Altitude

Year	Density (kg/m^3)		
	Day	Night	Average
1959 (0)	1.10×10^{-12}	2.40×10^{-13}	8.62×10^{-13}
1960 (1)	6.80×10^{-13}	7.30×10^{-14}	5.11×10^{-13}
1961 (2)	2.00×10^{-13}	3.10×10^{-14}	1.53×10^{-13}
1962 (3)	7.30×10^{-14}	1.90×10^{-14}	5.81×10^{-14}

Table III-B.4 Atmospheric Density at 700 km Altitude

Year	Density (kg/m^3)		
	Day	Night	Average
1959 (0)	4.90×10^{-13}	5.00×10^{-14}	3.67×10^{-13}
1960 (1)	2.00×10^{-13}	1.80×10^{-14}	1.49×10^{-13}
1961 (2)	5.00×10^{-14}	1.10×10^{-14}	3.92×10^{-14}
1962 (3)	2.00×10^{-14}	6.00×10^{-15}	1.61×10^{-14}

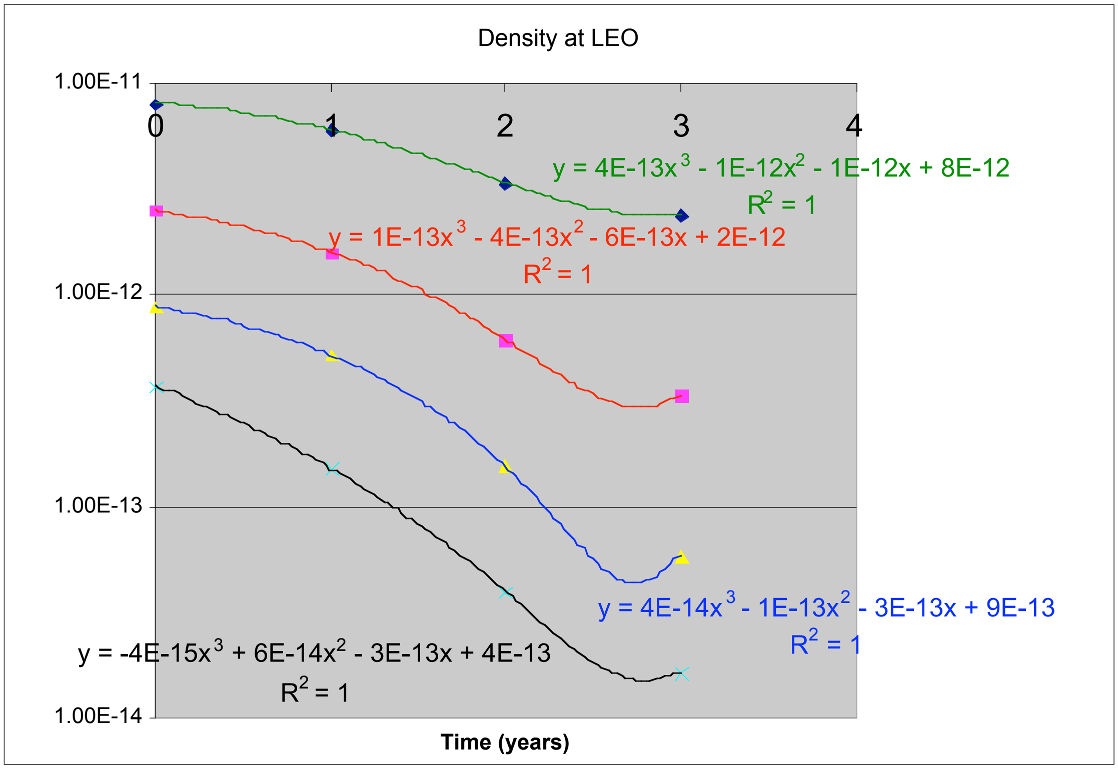

Figure III-B.1 Atmospheric Density (Average) Over Time

With such an extremely low density, the expected drag level is small. However, the orbital velocity of MicroMaps could range from 7.8 to 7.5 km/s (200 to 700 km), and drag is proportional to the square of velocity as opposed to the density which only has a linear proportionality to drag. The total impulse, the force of drag over a known period of time, is what will determine how much fuel is needed for the mission which is projected to last from 3 to 5 years. Table III-B.5 shows the values of velocity and total impulse at different altitudes assuming an average cross-sectional area of 1 m^2, an average drag coefficient of 4, and an average density over solar and day cycles.

Table III-B.5 Total Impulse and Orbital Velocity of MicroMaps

Altitude (km)	Velocity (km/s)	3 Year Impulse (kNs)
200	7.784	2,912
300	7.726	217
400	7.669	57
500	7.613	15

As seen from Tables III-B.1 through III-B.5, not only does atmospheric density decrease with altitude but also orbital velocity decreases with altitude. These facts are why the total mission impulse goes from approximately 2,900 kNs to 15 kNs by just increasing the altitude from 200 to 500 km. Recall from Section II-B that the 200 and 300 km orbits will decay to re-entry before the instrument life is up, while above 300 km the instrument life is reached before re-entry. The problem with increasing the altitude for the MicroMaps orbit is at 800 km, the craft begins to enter the Van Allen Belt, a region inside the Earth's magnetosphere where radiation is trapped. This radiation causes failure in electronics, loss in solar cell efficiency, and weakens metals, and the radiation increases with altitude. Even if the spacecraft takes a lower orbit, whenever the spacecraft flies over the South Atlantic Ocean region, it will fly through the South Atlantic Anomaly. The South Atlantic Anomaly is a potion of the Van Allen Belt that dips lower than the rest of the Belt. Thus, conflicting requirements are present. If the mission is designed for lower altitude orbits, a larger rate of orbital decay will be experienced and requirements on the propulsion system become necessary.

For low-level long-duration thrust requirements to cancel atmospheric and/or gravitational perturbations, the propulsion system of choice is electric propulsion.[12] Electric propulsion is based on one of three principles:

- Electrothermal - Thrust produced by electric power which heats a working fluid that expands out a nozzle,
- Electrostatic - Thrust produced by charged particles accelerated by an electrostatic field, and
- Electromagnetic - Thrust produced by the interaction of plasma with electric and magnetic fields.

There are two kinds of electrothermal systems: arcjet and resistojet. The resistojet uses a component with a high electrical resistance to heat the fluid. Arcjets uses an electric arc to generate heat. Although both produce a specific impulse higher than most chemical rockets, it is still lower than other forms of electric rocket propulsion. Also, their specific impulse decreases with the amount of thrust they produce. Therefore, thrust maneuvers would not be constant but instead consist of periodic high thrust level orbital reboosts.

There is only one kind of electrostatic rocket, the ion rocket. Ion rockets have been researched since the 1950's and have seen extensive use over the last decade for station keeping for geostationary satellites and even for a recent NASA space probe to an unexplored comet (Deep Space I). These rockets generate the low thrust/high specific impulse needed for long duration spaceflight requiring a counter balance to atmospheric drag. However, like the electrothermal rockets, they loose efficiency and specific impulse if the thrust level is too low. A power source must be at least 100 W for it to be effective. Also, they generate a large positive charge on the ship that must be countered by "bleeding off" the negative charge and applying it to the positive ion engine exhaust.

Electromagnet propulsion propels plasma using interactions between the plasma and magnetic and electric fields. Although plasma is composed of charged particles, the net sum of the charge of a volume of plasma is zero, thus avoiding the problem of the engine building up a charge. There are three different types of electromagnetic engines: magnetoplasmadynamic, Hall thruster, and solid pulse plasma. The magnetohydrodynamic engine will not be discussed since the technology is still in the development stage. The Hall thruster has a specific impulse of 1,500-5,000 s, relatively simple electrical system, uses inert xenon and argon as fuel, and the technology has been used in approximately 100 Russian satellites over the past 20 years. However, like the ion engine, the power consumption is high and near 100 W. There is also the problem of beam divergence, and engine erosion from the exhaust. The other electromagnetic propulsion system is solid pulsed plasma engine, also known as the pulse plasma thruster (PPT). This engine consists of a block of teflon that is ablated off with a high electric current forming a plasma which is accelerated across the nozzle. The process of ablating the teflon fuel into a plasma is done on discreet intervals, hence the term "pulse plasma". PPT is the simplest of the three electromagnetic engines to use since the fuel and rocket are integrated together in one engine instead of a separate tank for fuel and eliminating the need for a fuel feed system. The power requirement is a minimal 100 W. However, since the system is a pulsed system, capacitors can store charges for a short time and release it for the engine, lowering the power requirement. However, the teflon fuel is not user friendly. Teflon is toxic, corrodes the nozzle, and will collect on the nozzle lining over time. Also, the PPT has the lowest efficiency of any of the electromagnetic engines. Also, the PPT has just been recently developed and does not have the technical maturity of the Hall thruster. Only 10% of the power put into the PPT goes into the thrust leaving the engine at an average specific impulse of 550 s.

Table III-B.6 is a chart rating each electric rocket propulsion system on several criteria. According to the study, the best engine to use overall is the solid pulse plasma engine followed by the Hall thruster. The largest design criteria was whether or not the engine produced long duration, low thrust. Long duration, low thrust settings were chosen as opposed to high thrust burns for reboost because it was easier to perform satellite tracking if the thrust was just enough to cancel out atmospheric drag. Also, high thrust reboosts could hinder observations made by the satellite.

Table III-B.6 Electric Propulsion Performance Chart

Engine Type	Long Duration Low Thrust	Power Consumption	Fuel Requirements	Simplicity Reliability	Space Heritage
Resistojet	□	√	□	√	√
Arcjet	□	√	□	□	□
Ion Engine	√	□	√	□	√
PPT	√	√	√	□	□
MPD	√	□	√	√	□
Hall Thruster	√	□	√	□	√

√ high rating, □ medium rating, □ low rating

After doing an information search on the internet for companies providing electric rockets and contacting them by email asking for information on their product, the choices for the engine was narrowed down to two. The two choices are the CU Aerospace PPT-8/9 and the Busek Tandem-200.[13,14] Table III-B.7 is a chart for each engine's performance, fuel requirements, and cost. The first rocket engine, the Tandem-200, is a Hall thruster made by Busek. This system has a higher specific impulse and thrust and larger range of operating power than the PPT-8/9 by CU Aerospace. The Tandem-200 has a higher power requirement, higher thruster mass, and need of a fuel tank and feed system. Also, the cost of xenon is higher than gold, while the fuel for the PPT-8/9 is included inside the engine at a lower value. The cost of the PPT-8/9 is $30,000 per thruster. The thrusters are integrated into a set, up to eight, and used together. The electronics and power source can be operated in series allowing the thrusters to use the same power and electronics for operation. Thrusters are placed together and fired one at a time to avoid unbalanced thrust. The price of the Tandem-200 was quoted to be several hundred thousand dollars for the engines, electronics, and power source. Due to the simplicity of integration and use, the PPT-8/9 is preferred over the Tandem-200. However, the deciding factor will be the orbit of the spacecraft.

Table III-B.7 Candidate Electric Rocket Engines

System	Tandem-200	PPT-8/9
Company	Busek	CU Aerospace
Type	Hall Thruster	Pulse Plasma Thruster
Thrust (mN)	12.4	2.9
Total Impulse (Ns)	15,680/kg of fuel	1,225
Specific Impulse (s)	1600	550
Power Consumption Nominal (W)	200	120
Power Consumption Range (W)	50-300	100-150
Mass (kg)	0.9	0.4 (with integrated fuel)
Mass Flow Rate Nominal (mg/s)	0.94	0.538
Fuel	Xenon	Teflon
Cost	$100,000 to $1 M	$30,000 per unit $500,000 electronics

Table III-B.8 is a chart of the amount of fuel for the Tandem-200 and the number of PPT-8/9 units needed for a three year mission. As seen here, the PPT-8/9 becomes impractical under an altitude of 400 km based on the required number of units. Only the Tandem-200 could produce a high enough total impulse over the duration of the mission. However, at extremely low altitudes (200 km), the required fuel mass also becomes impractical.

Table III-B.8 Mission Requirements for the Tandem-200 and PPT-8/9 Engines

Altitude (km)	3 Year Impulse (kNs)	HT Fuel Mass (kg)	Number of PPT Units
200	2,912	185	2,377
300	217	13.8	177
400	57	3.6	46
500	15	1.0	12

C. Electrical Power Generation and Storage

In all likelihood, power generation for the MicroMaps small dedicated space platform will be implemented with solar energy conversion. The solar photovoltaic cells for the MicroMaps satellite will be placed along the surface of the satellite. Consequently, the satellite will most likely be hexagonal-shaped. Because the satellite must keep the CO measurement instrument facing the Earth in the nadir direction at all times, the Sun will shift its position relative to the satellite. Photovoltaic cells must be at a perpendicular angle to its source of light for the maximum amount of light to be absorbed and converted to electricity. This observation is why a hexagonal-shape is expected for the satellite's structure. As the Sun shifts position in relation to the satellite, there will always be enough photovoltaic cells in the proper position to produce the necessary amount of electricity.

The needed solar cell surface collection area is dependent on the amount of time the Sun is shining on the satellite, the amount of power used, and the percentage of solar light converted to electricity. The amount of time the satellite is exposed to the Sun depends on its orbit. Higher altitude orbits allow for an increase in the length of the daylight for the satellite. However, since the range of orbits is only between 200 to 1,000 km in altitude, there are only marginal changes in the amount of sunlight received over this range. Figure III-C.1 is a graph of the percentage of daylight a satellite experiences in a 700 km orbit at a high inclination angle for the period from January 1, 2004 to December 31, 2004. From the graph, it can be seen that even during the days with the shortest periods of sunlight, the satellite still receives daylight for 70% of the time and some days experience a full 24 hr of continuous sunlight. For the minimally lighted days, this behavior would leave a period of 7.2 hr without sunlight.

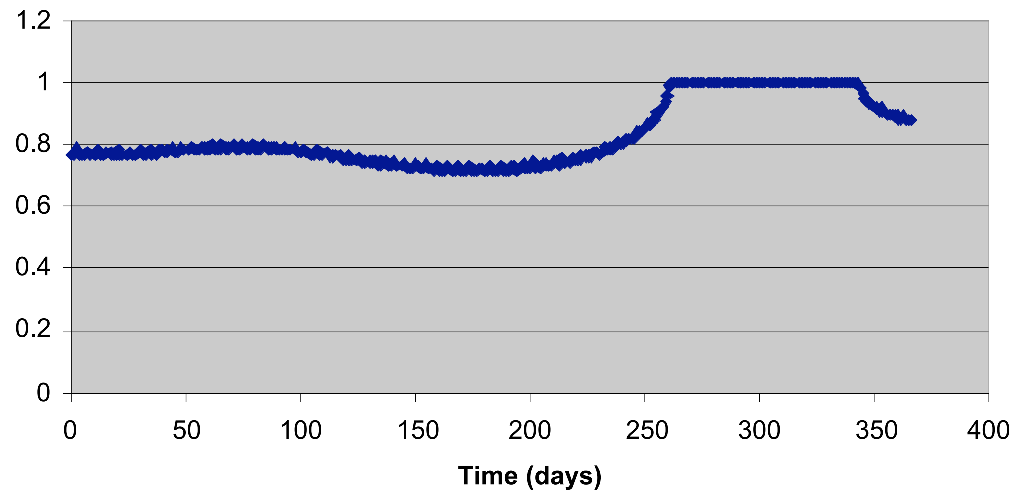

Figure III-C.1 Typical Low Earth Orbit Spacecraft Lighting Cycle

The amount of power needed for MicroMaps is estimated at 300 W including a minimum of 100 W for the propulsion system. Also, the solar cells must generate an additional 135 W for charging the nickel-cadmium batteries used during periods without sunlight. The highest conversion rate for a satellite solar cell[15] is approximately 26.5% and the intensity of sunlight at Earth's orbit is 1,371 W/m^2. The area of solar panels needed for powering the spacecraft is thus about 1.2 m^2. Of course, this is the area of solar cells exposed to direct sunlight at a perpendicular angle to the Sun. Thus, the actual surface area must be much larger than 1.2 m^2. Surface area estimation of the satellite is difficult to say at this stage without further studies of satellite configuration design and orientation in relations to the Sun. There are also cost and weight restraints to deal with. Nickel-cadmium batteries have a specific energy, the amount of energy that is stored per unit mass, of 219 W-hr/kg. In order to provide 300 W for over seven hours of darkness, there must be 10 kg of NiCd batteries on board the satellite. The estimated mass of the satellite is 50 kg. Battery mass alone would be 20% of the satellite's mass. Also, fully integrated photovoltaic cells are estimated to cost 700 $/W.[15] For MicroMaps, the cost would be over $300,000. The dollar amount will be several times larger since there will be more than 1.2 m^2 of satellite surface area to be covered in photovoltaic cells. Another option is to place a smaller area of solar cells on movable panels. However, this would cause complexities in attitude control, vibrations through the satellite creating jitter motions in the instrument observations and corrupting the measurement data. In order to avoid this, the control system would have to be more complex, causing added cost to the mission. Finally, articulating panels would increase satellite drag and the fuel, mass, and power needed for the propulsion system in low altitude orbits.

In order to reduce the power requirements for the satellite, the propulsion system could be left off during days with the shortest periods of sunlight, or during times without any sunlight. By just keeping the propulsion system off during the periods of time without sunlight, NiCd battery mass could be reduced to 6.6 kg. Requiring that at least 80% of the day the satellite is in sunlight before operating the propulsion system and only operating during the day, the effective area needed for the photovoltaic cells would only be 0.95 m^2. Power management will be very important for the mission to save cost, mass, and system complexity.

A detailed power consumption estimate for the Attitude Determination and Control System is presented here. The reaction wheels at peak operating conditions will consume approximately 25 W each, and at steady state they will consume 7 W each. If one reaction wheel is considered operating at its peak value while the other three are in steady state (which is achievable with a properly designed control system), the total power required for the four reaction wheels would be $25 + 7 \times 3 = 46$ W. Torque rods rated at 1 Am^2 used for dumping the excess momentum from the reaction wheels will consume about 0.28 W of power. For 3 torque rods, around 1 W is needed. Such torque rods will make the dumping process very slow. If higher capacity torque rods of 12 Am^2 are used which consume 0.7 W each, a total of about 2 W is needed. The magnetometer needs around 1 W of power for operation. Sun and Earth sensors consume around 2 W each, amounting to a total of 8 W. Therefore, an estimate for total power consumed by the ADCS is estimated to be approximately 60 W.

D. Vehicle-Ground Communication and Telemetry

Communication between the proposed spacecraft and the ground stations will at a minimum consist of the following components.

- MicroMaps Data Downlink
- Camera Data Downlink
- Spacecraft Telemetry Downlink
- Ground Command Uplink

Sizing the communication system will affect the overall sizing of the satellite and will also be affected by some key system drivers. This subsection briefly presents some of the system drivers that affect communication system size.

The MicroMaps data is generated and compressed inside the instrument itself. The data rate coming from the instrument is 0.432 Mbyte/day, which is 40 bit/s. The telemetry data rate is not yet known, however 30 kbit/s is taken as a start point. The command data rate is not yet known, however 3 kbit/s is taken as a start point. The camera imaging data rate is not yet known, however 25 Mbit/s is taken as a start point for the imaging rate. An assumption made here is that one ground station is available to receive MicroMaps data. As the satellite flies around the Earth, MicroMaps will always collect data and store it in mass memory onboard. The satellite will downlink data each time the ground station is available. The satellite collects a complete set of data for the whole Earth in m days, the revisit time period from Section II-C. The satellite is required to downlink the complete set of data for the whole Earth in m days also. Total MicroMaps data stored in m days equals 3.3m Mbits.

A gain shaped antenna can be used to cover the whole horizon under the satellite with elevation angle of 5 deg. Four different orbits are investigated. For each orbit, the beam width of the antenna is calculated and the time available for downlink of MicroMaps data, T, is also calculated. From that, the bit rate for downlink is calculated for MicroMaps data. Results are listed in Table III-D.1 and Figure III-D.1. From the computed values of the downlink bit rate, one can use a single set of transmitter and antenna components for both telemetry and MicroMaps data. Either UHF or S band frequencies can be used. Both frequency bands may be used for redundancy. To get an estimate for the required memory for MicroMaps data, notice that the maximum period for the satellite in which it cannot see the ground station is two days; the required mass memory is thus $2 \times 24 \times 3{,}600 \times 40 = 6.6$ Mbits. The downlink bit rate affects the power consumption of the communication system. Typical behavior for the variation of transmitter power with the bit rate is plotted below in Figure III-D.2.

Table III-D.1 MicroMap Data Downlink Bit Rate

Altitude (km)	m (day)	Beam Width (deg)	T (min)	Data Rate (kbit/s)
461	20	136	93.8	12.0
542	25	133	174.2	8.1
676	23	129	195.4	6.6
776	26	125	257.4	5.7

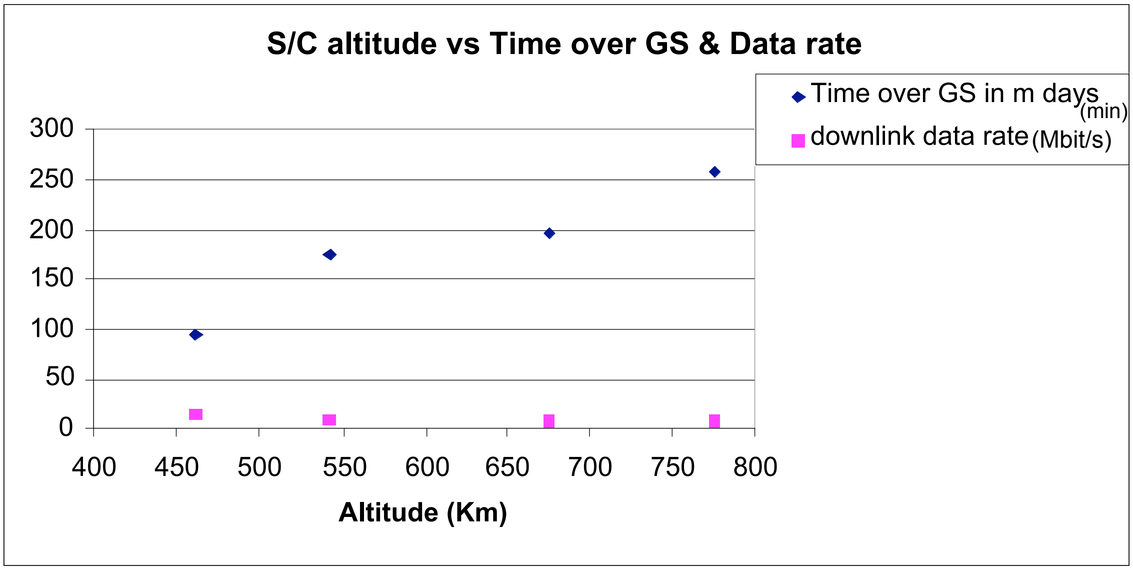

Figure III-D.1 Altitude vs. Ground Station Time and Data Rate

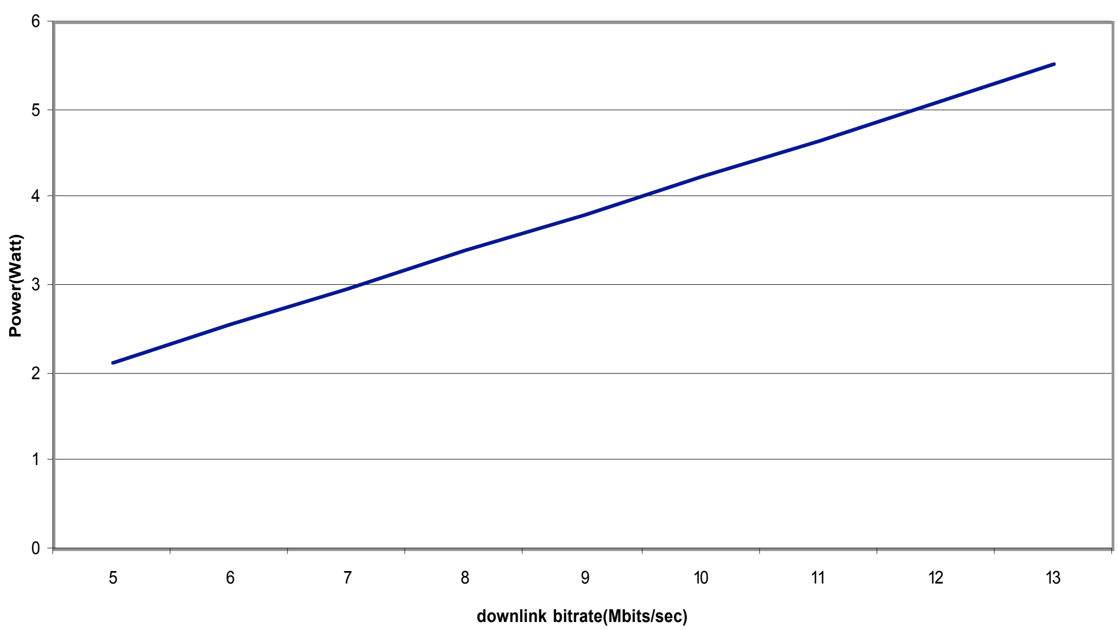

Figure III-D.2 Power Consumption vs. Downlink Bit Rate

E. Earth Observation Camera

Camera system functions are to correlate images of major sources and sinks of CO with the MicroMaps CO measurement. Resolution of the Earth images is uncertain at this time. The components of the camera payload are envisioned to include the following items.

- Telescope
- Detectors
- Signal Processing

Sizing the camera system will affect the overall sizing of the communication and telemetry data stream components. This subsection briefly presents some of the system drivers that affect the camera system.

The telescope is responsible of collecting the reflected sunrays from the Earth surface and focus it in the focal plane (detectors plane). Thus, the telescope is controlling the swath width with the number of detectors, as well as brightness of the image which is the amount of light that passes to the detectors. The two main parameters that specify the telescope performance are aperture diameter (D) and focal length (f), which leads to the calculation of the F number (F#).

$$F\# = f/D \qquad \text{(III-E.1)}$$

The F# gives an indication of the optical quality of the image, the smaller the F# the better the image. The minimum F# is 0.5 (i.e., max D = 2f). Typical range values for F# are from 4 to 6.

No matter how good the quality of the lens or mirror, the main limitation to resolution is diffraction. This constraint is caused by the diffraction of the incident rays at the edge of the aperture of the telescope, which leads to the generation of constructive and destructive interference patterns on the image plane, as can be seen from Figure III-E.1. This phenomenon limits the resolution to

$$\theta_r = 1.22\,\lambda/D \qquad \text{(III-E.2)}$$

This limitation leads to what is called the quality factor (Q). Quality factor is defined as the ratio of pixel size (d) to the point spread function (d'), or

$$Q = d / d' \qquad \text{(III-E.3)}$$

To have a suitable image quality, Q must be greater than 1, so in our design the quality factor will be assigned as Q = 1.1. This fact can be seen from Figure III-E.2. Thus, the assigned value for Q will achieve the required optimization between the small data rate and high ground resolution.

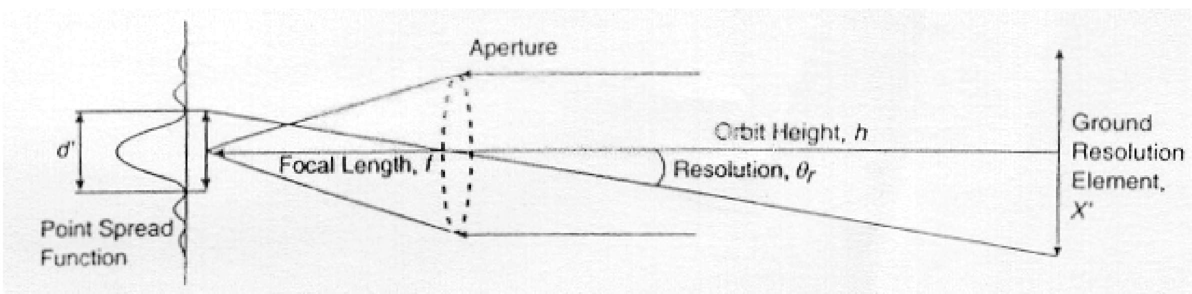

Figure III-E.1 Lens Diffraction Illustration

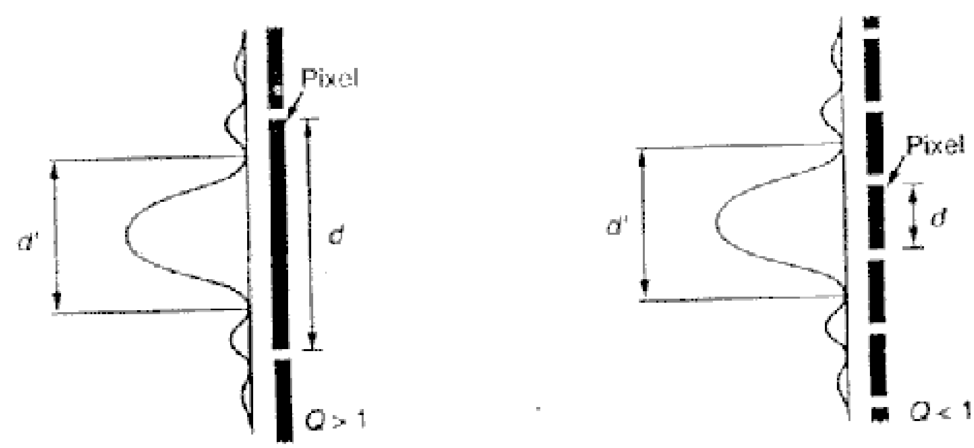

Figure III-E.2 Image Quality Geometry

In designing the detecting system one has to deal with the swapping technique and the type of detector. The swapping technique decides the style of the detector matrix that one is going to use. The main types of swapping techniques include

- Single Element Whiskbroom Sensor
- Multi-Element Whiskbroom Sensor
- Push Broom Sensor
- Matrix Imager

These various types are illustrated in Figure III-E.3. The recommended type for use is the push broom because it combines the advantage of acceptable picture quality with less complexity.

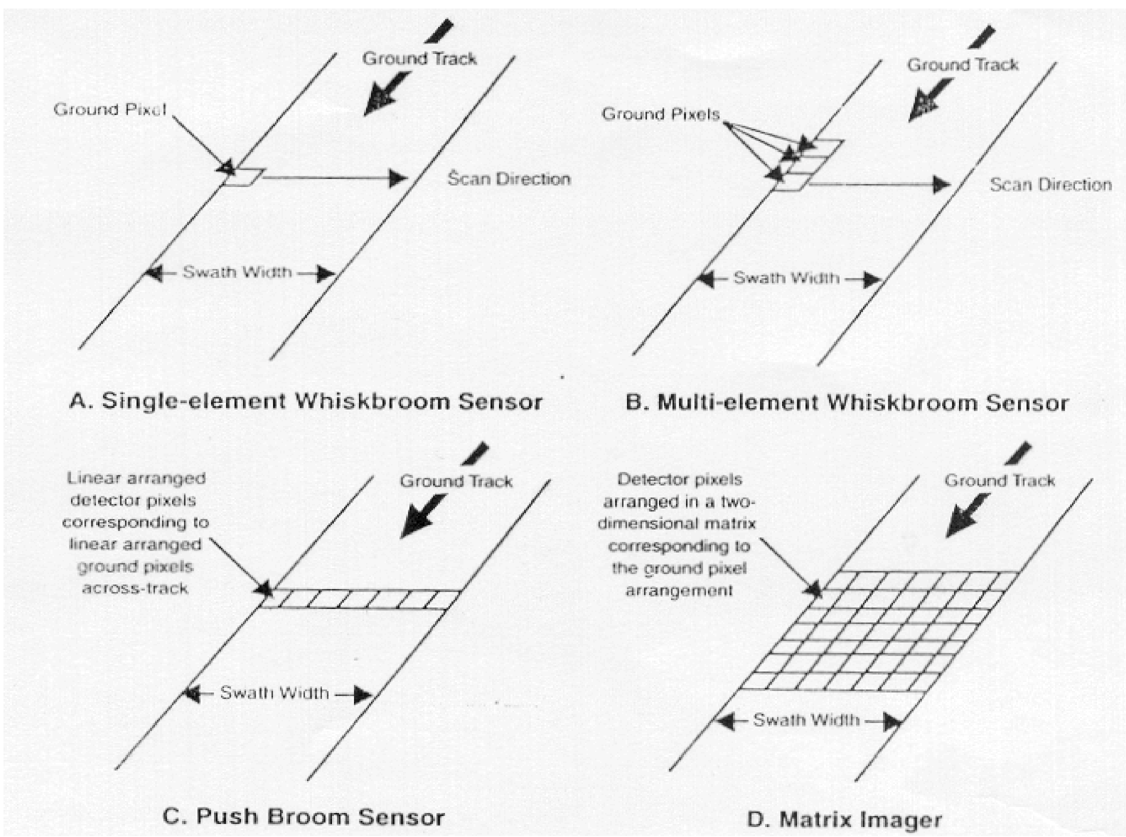

Figure III-E.3 Swapping Technique Options

The next stage of analysis addresses pixel depth which affects image quality and bit rate dramatically. Pixel depth (B) is defined as the number of bits that represent each pixel. In this design study, pixel depth will be assigned as B = 9 bit. This assignment can give a total number of colors equal to 512, which is more than acceptable. This value means that each color (near IR, red, and green) are assigned 3 bits each. This value also simplifies the electronics in the payload by avoiding generation of chromatic signals and then converting them. Here each color is converted separately. This pixel design also achieves good quality at the same time by reducing noise and error that arises from combining the three basic colors together first and then trying to re-extract them again. This pixel depth also allows high flexibility in compression techniques by leaving the selection choice either to compress each color by itself or combine them together first.

To calculate the design parameters of the camera system that match the satellite orbit and the required picture quality, a multi-step calculation is considered. Input data to this analysis includes the following values provided by the mission analysis.

- Earth Radius (Re) = 6378.14 km
- Orbit Altitude (H) = 673 km
- Orbit Period (p) = 98 min
- Ground Track Velocity (Vg) = 6.815 km/s
- Required Ground Resolution (X) = 20 m
- Required Swath Width (X_{max}) = 40 km

Output data from this analysis includes the following information.

- Aperture Diameter (D): (width of telescope lens)
- Focal Length (f): (distant between lens and image sensor)
- Data Rate (DR): (data transfer rate to download images instantaneously)
- Number of Detectors (Z_c): (Number of image sensors needed to cover required swath)
- Pixel Width (d): (width of image sensor)
- F Number (F#) : (measure of optical system efficiency and design, lower F# is desirable, typical value is 4 to 6)

Design parameters used in this analysis are also listed below.

- Pixel Depth (B) = 9 bit
- Image Quality Factor (Q) = 1.1

Finally, the image sensor model assumes a push broom configuration with 3 lines of spectral sensors using the following wavelengths.

- λ_g = 550 nm (green)
- λ_r = 650 nm (red)
- λ_{IR} = 750 nm (IR) = λ_{max}

The camera system analysis uses the Earth, satellite, and destination point geometry illustrated in Figure III-E.4.

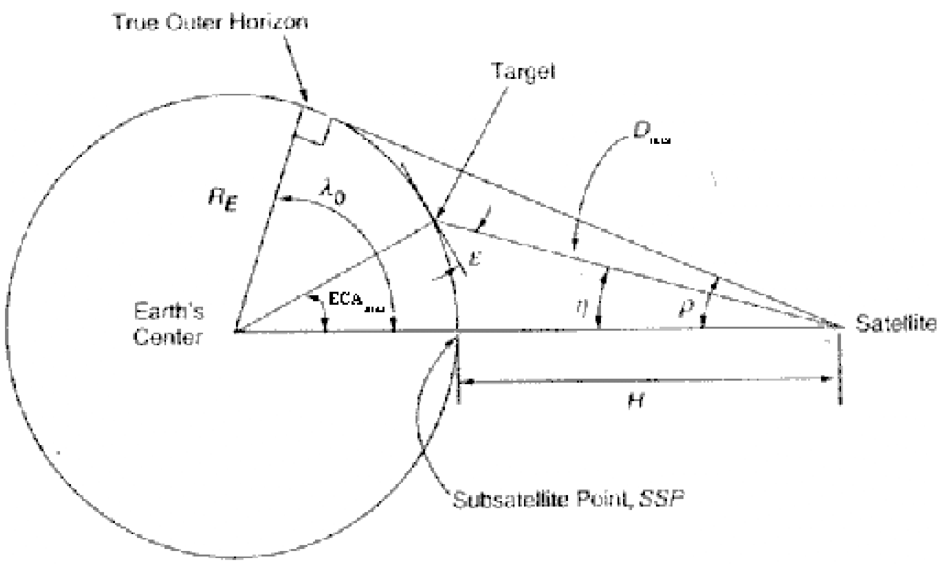

Figure III-E.4 Camera Viewing Geometry

The initial set of calculations determines the required number of detectors.

- Earth Angular Radius (ρ) = $\sin^{-1}(Re/(Re+H))$ = 64.76 deg
- Max Central Angle (ECA_{max}) = (swath width) / 2Re $\times$ 180/π = 0.18 deg
- Min Elevation Angle (ε) = $\tan^{-1}\{\cot(ECA_{max}) - \sin(\rho)/\sin(ECA_{max})\}$ = 88.12 deg
- Sensor Look or Nadir Angle (η) = 90 - ECA_{max} - ε = 1.7 deg
- Max View Distance (D_{max}) = H /cos(η) = 673.3 km
- Instant Field of View (IFOV) = $2\tan^{-1}$ (Ground Resolution / $2D_{max}$) = 0.001702 deg
- No. of Pixels in Cross Track Axis (Z_c) = 2η / IFOV = 2,000 pixels (single color)

The next set of calculations determine the data rate.

- No. of Swath Records per Second (Z_a) = Vg $\times$ s / X = 340.75 sr/s

- No. of Pixel Records per Second (Z) = Z_a □ Z_c = 681,500 pr/s
- Data Rate (DR) = Z □ B = 6.1335 Mbit/s (single color), DR = 2.3 Mbyte/s (3 colors)

The final set of calculations determines the optical system parameters. A typical value for the detector width is d = 7 □m. For example, the NEC □pd3799 or NEC □pd3798 commercial units both can be used in the MicroMaps Mission as liner CCD image sensors.

- Focal Length (f) = h □ d / X = 23.6 cm
- Aperture Diameter (D) = 2.44 □ $□_{max}$ □ Q □ h / X = 6.75 cm
- F Number (F#) = f / D = 3.5

A parametric study can also be conducted using an Excel spreadsheet to uncover the effect of each of the above parameters. Figure III-E.5 illustrates the variation of the imaging data rate vs. the resolution for different values of the camera swath width. This chart is useful for deciding whether to look for global image coverage of the Earth or accept a certain coverage percentage.

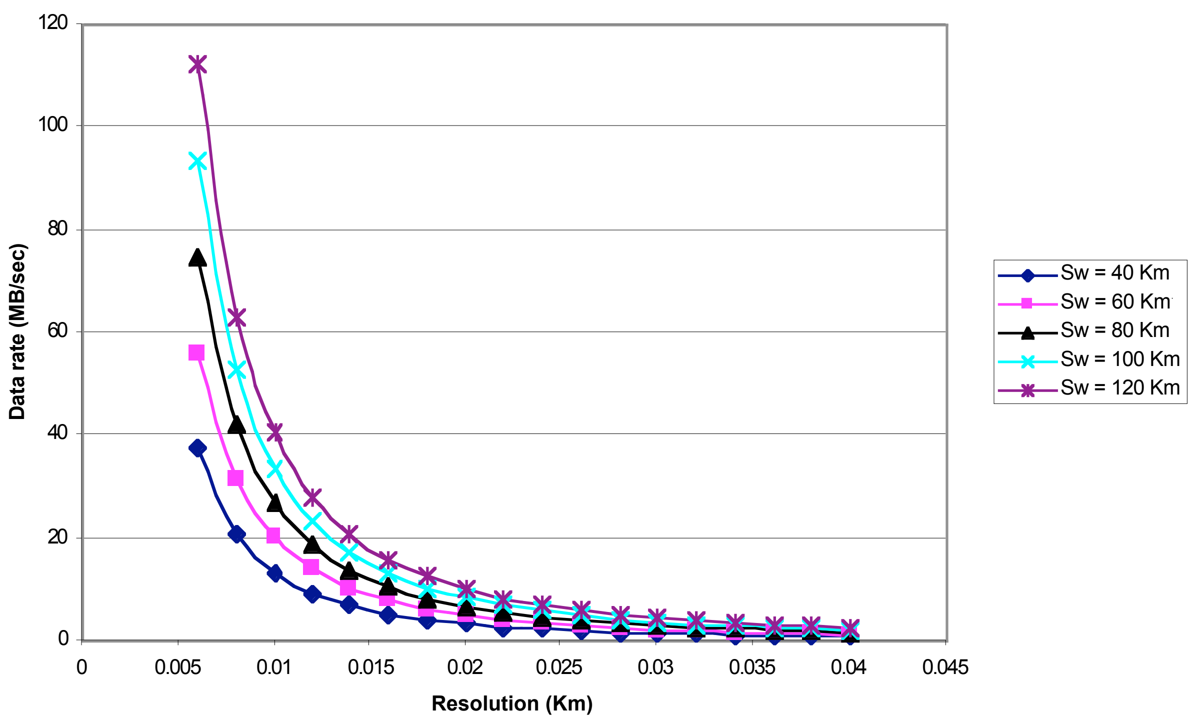

Figure III-E.5 Data Rate vs. Resolution (Constant Swath Width)

Not only the data rate is affected by varying the swath of the camera for a certain resolution, but also the size of the camera itself, and of course its cost, will be affected. One can get a sense of how the camera size will increase by calculating the required number of detectors. Figure III-E.6 illustrates the variation of the number of detectors for one color by varying the camera swath and the resolution.

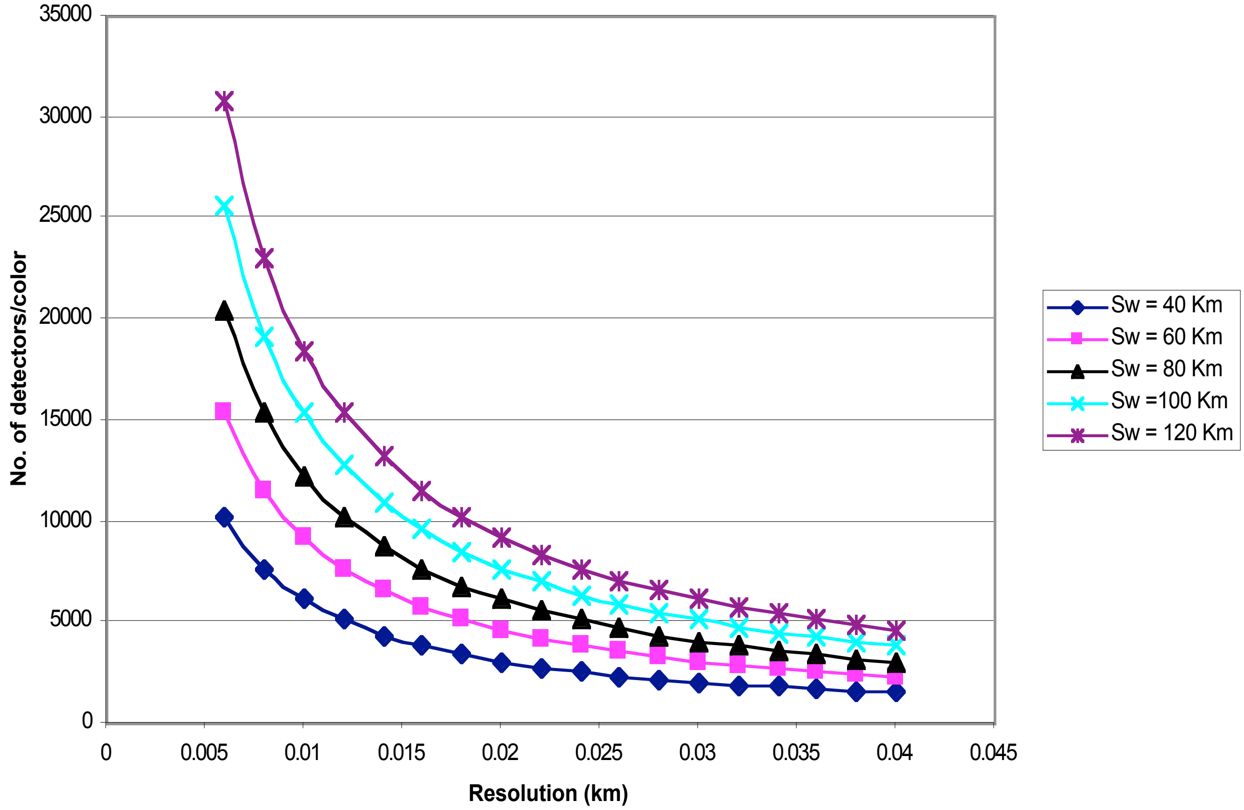

Figure III-E.6 Number of Detectors vs. Resolution (Constant Swath Width)

Section IV
International Space Station - Key Issues

A. Earth Surface Coverage

The International Space Station orbit is shown in Figure IV-A.1. This orbit has an altitude of 407 km and an inclination of 51.6 deg. Clearly observe from this figure that the ground track associated with this orbit covers a large portion of the terrain of the Earth. Mission planners have suggested that the Space Station can be used as a suitable platform for the MicroMaps Mission. One advantage of using this platform is the possibility of riding on the Space Shuttle as a launch vehicle during one of the scheduled missions to the Space Station. This approach solves the launch vehicle opportunity problem. Yet, problems associated with this option might arise from the unknown behavior, at least to the mission planning team at this time, of the Space Station platform. Another advantage of flying on the Space Station is the possible elimination of the task of designing a satellite to carry the instrument. However, such an effort would detract from the objective of developing and enhancing student technical capabilities and educational experience in the area of engineering support for space systems.

The Space Station orbit completely covers the land in the southern hemisphere, and all land south of latitude 51.6 deg in the northern hemisphere. Unfortunately, all of Russia, the most part of Canada, northern parts of Europe, Alaska, and the polar regions are not covered. This deficiency in Earth surface coverage can be observed from Figure IV-A.1. This behavior means that if a significant CO source or sink is positioned in any of these regions, its characteristics will not be directly detected by MicroMaps. Yet, because of global scale atmospheric air motion, weather, wind, etc., indirect effects from this CO source or sink would be detected elsewhere in the covered regions. Incomplete Earth surface coverage can lead to uncertainty in global atmospheric models and climate projections. Geographic regions covered by the Space Station orbit were previously studied during Space Shuttle MAPS missions.[2,3] Measurement of the polar

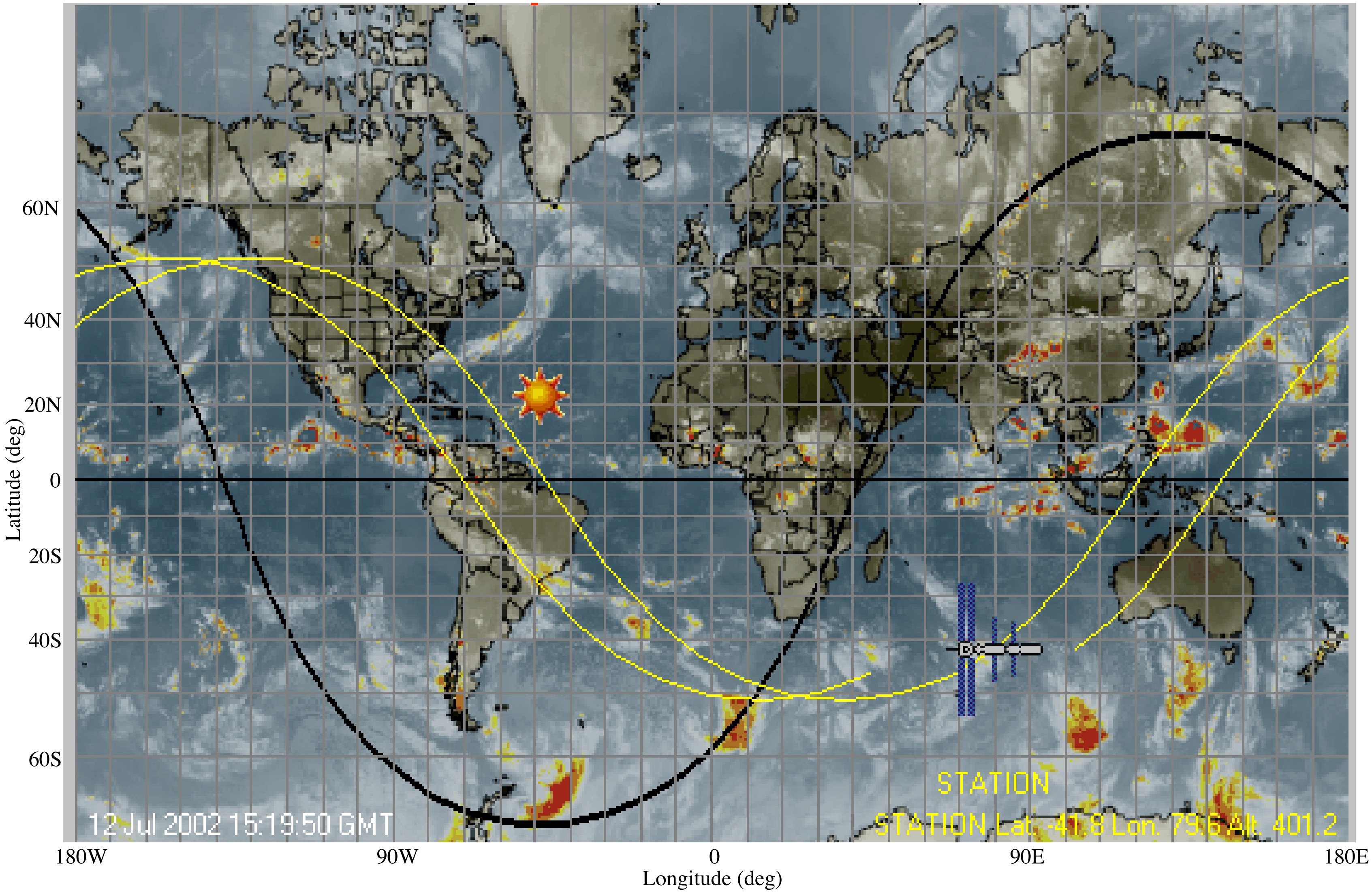

Figure IV-A.1 International Space Station Earth Surface Coverage regions will be out of the question because the orbit does not even come close to these latitudes. A complete global map of CO distribution will not be possible, only a partial map between latitudes 51.6 deg north and south will be possible.

As documented in Section II-B, scientific objectives highly emphasized a global CO distribution measurement as opposed to just covering lower latitudes. Mission scientists underscored this objective on several occasions. Therefore, based on orbit suitability and associated Earth surface coverage, flying MicroMaps on the International Space Station platform is not recommended. This option should be discarded.

B. Attitude and Vibration Transients

One of the major roles the International Space Station will fulfill is to serve as a multi-user platform for long term atmospheric, ocean, land, and astronomical scientific investigation. Additionally, exploitation of the microgravity and/or vacuum space environment for scientific and commercial purposes is expected. Unfortunately, the Space Station will be a dynamic platform that experiences attitude and vibrational motion transients originating from a multitude of operational constraints that may corrupt or compromise the user requirements depending upon the application. Evaluating the Space Station attitude and vibrational dynamic characteristics against the MicroMaps requirements will therefore be addressed in this subsection.

Figure IV-B.1 shows the fully operational Space Station configuration.[16] The vehicle is characterized by a long slender truss structure serving as a backbone with numerous facilities, modules, and solar arrays attached along its length. The span of this truss structure is approximately 108 m while the transverse attachments are about 80 m long. Users will attach hardware to available pallets that are located along the truss structure. These pallets are oriented in both the +Z and -Z directions, and can be located a significant distance from the vehicle mass center. The Space Station will be flown in several operational modes with varying orientations, the solar panels will be actively articulated for optimum solar tracking, robotic arms and track

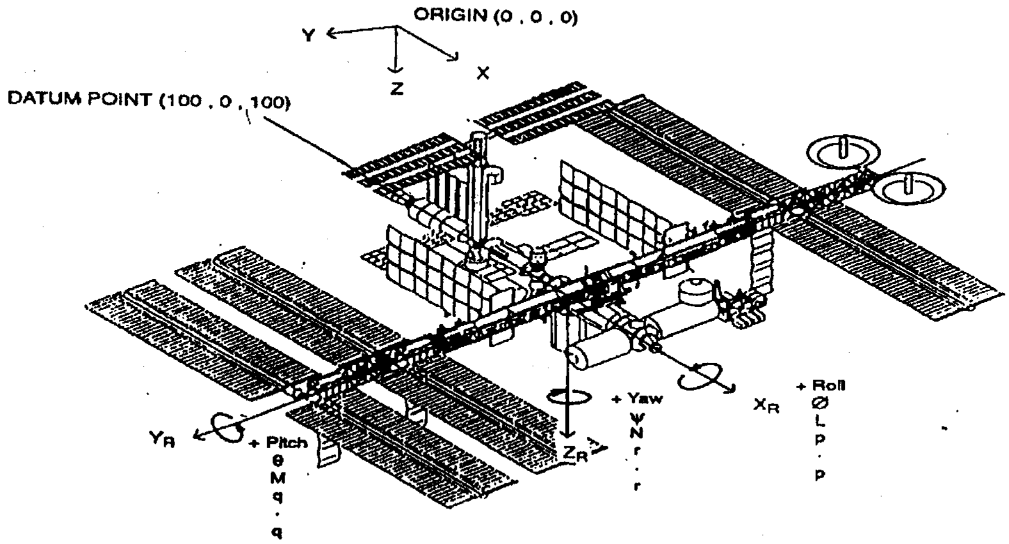

Figure IV-B.1 International Space Station Configuration

vehicles will be performing construction and maintenance duties, service and supply vehicles will be docking frequently, periodic orbit boost maneuvers will be executed, angular momentum control devices will be in operation, and the vehicle is a large light weight flexible structure susceptible to disturbance propagation. In addition, the current configuration will undergo many on-orbit modifications over the next several years before achieving the fully operational configuration of Figure IV-B.1. These configuration modifications encompass large changes in inertia and attitude control capability. In summary, the International Space Station has a potential for exhibiting significant attitude and vibration transients which could compromise the scientific integrity of data collected by MicroMaps.

During the next several years, the Space Station will be operated in various flight attitude modes.[17] These modes are summarized in Table IV-B.1 and Figure IV-B.2. Mode XVV is a flight attitude where the X axis is near the Velocity Vector. This mode minimizes aerodynamic drag and is used to achieve microgravity conditions and for orbit boost maneuvers. Mode XPOP is a flight attitude where the principal X axis is Perpendicular to the Orbit Plane. This mode simultaneously provides for optimum solar collection and power generation and minimizes the gravity gradient torque. Mode TEA is a flight condition where environmental torques are in approximate balance, i.e., Torque Equilibrium Attitude. This mode balances aerodynamic torque and gravity gradient torque and is used to achieve microgravity conditions. Mode SSD is a flight attitude used for Space Shuttle Docking, while Mode SVD is a flight attitude used for Service Vehicle Docking.

Table IV-B.1 Space Station Flight Attitude Modes

Mode	Description	Yaw - Z (deg)	Pitch - Y (deg)	Roll - X (deg)
XVV	X Axis Near Velocity Vector Min Aero Drag, Microgravity, Orbit Boost	+ 15.0 – 15.0	+ 15.0 – 20.0	+ 15.0 – 20.0
XPOP	Xp Axis Perpendicular To Orbit Plane Min Gravity Torque, Max Solar Collection	+ 10.0 – 10.0	+180.0 –180.0	+ 10.0 – 10.0
TEA	Torque Equilibrium Attitude Aero-Gravity Torque Balance, Microgravity	+ 13.1 – 12.0	+ 2.8 – 19.1	+ 1.2 – 2.6
SSD	Space Shuttle Docking Shuttle Docking Procedures, Similar To XVV	+ 0.0 – 0.0	+ 0.0 – 0.0	+ 0.0 – 0.0
SVD	Service Vehicle Docking Service Docking Procedures, Similar To TEA	+ 0.0 – 0.0	+ 15.0 – 20.0	+ 15.0 – 20.0

The yaw-pitch-roll attitudes (referenced to a local level frame) listed in Table IV-B.1 indicate nominal operating ranges expected over the next several years as the platform configuration undergoes modification and expansion. For example, in the XVV mode, the pitch angle will lie somewhere between +15 and –20 deg, such as +10 deg, for an extended period of time on the scale of many months. After this extended period, the platform is modified and a new pitch angle results, such as –5 deg. This new angle remains until the next modification. The ranges listed in Table IV-B.1 should not be interpreted as bounding continuous transients occurring on an hourly-daily scale. The one exception to this interpretation is pitch angle range for the XPOP mode. The XPOP mode holds an inertial orientation, thus yielding a ±180 pitch angle variation during each orbit. Returning to the XVV mode +10 deg pitch angle example, the Space Station will not hold this precise attitude in the short term either. Due to previously listed disturbances, actuator and/or sensor hardware limitations, control performance, and structural

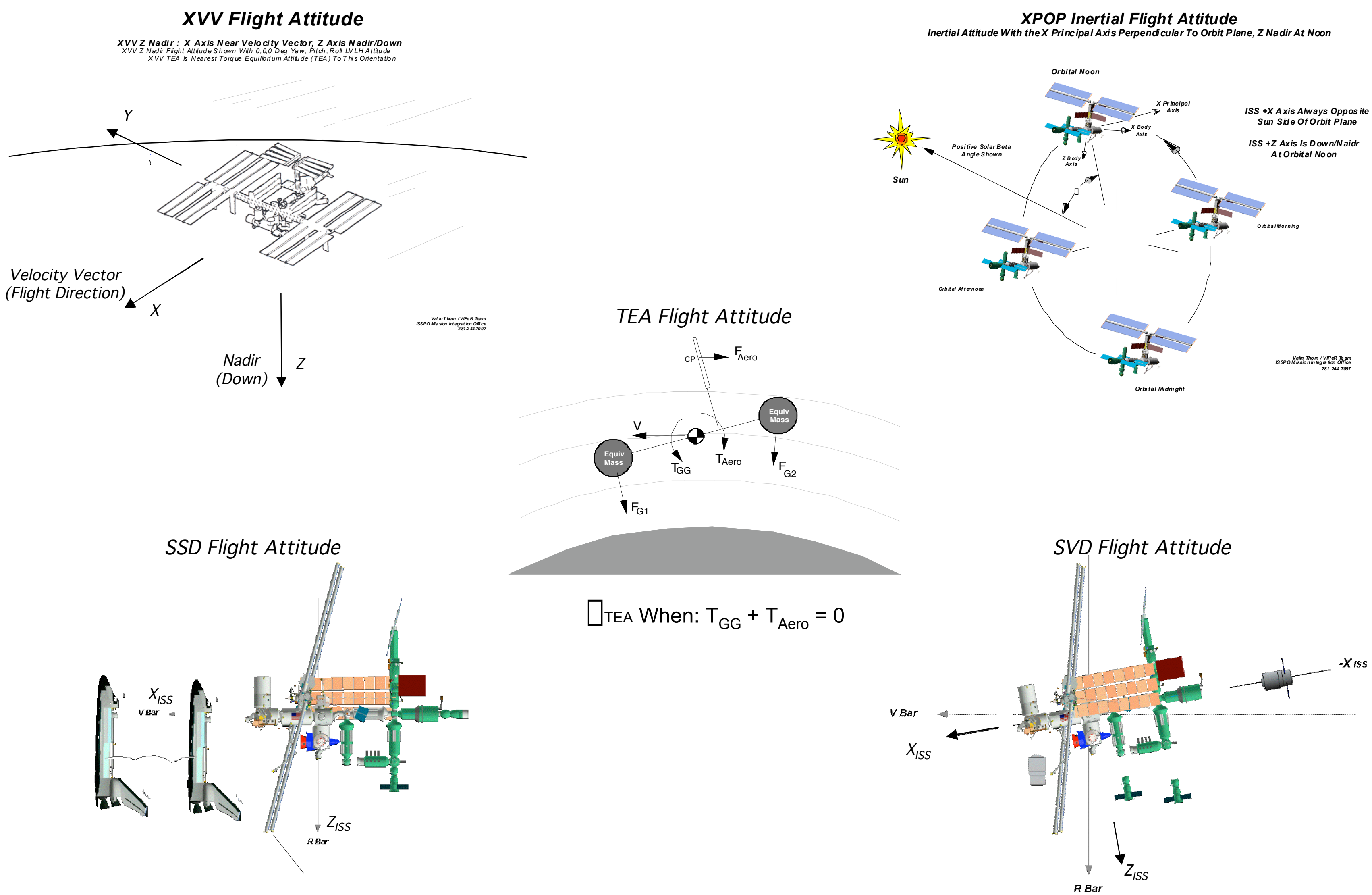

Figure IV-B.2 Space Station Flight Attitude Modes dynamics, small pitch transients will continuously occur around the nominal value. References 18-19 specify maximum transients to be ±3.5 to ±5 deg, depending upon platform configuration. Although the Space Station is a large flexible structure, the attitude control and disturbance bandwidths are well below the structural dynamic resonant frequencies. As shown in Reference 20, pointing disturbances due to structural vibrations are projected to minor.

Now interpret the information in Table IV-B.1 with respect to the MicroMaps Space Mission requirements. MicroMaps would be mounted to the Space Station such that the instrument would look along the +Z axis (see Figure IV-B.1) since +Z is approximately oriented along nadir in most cases. With this arrangement, only the pitch-roll variations in Table IV-B.1 are of concern. Yaw would not affect nadir viewing or measurements. Optimum pitch-roll values are 0 and 0 deg. Recall the nadir pointing accuracy requirements for MicroMaps from Section II-B: ± 5 deg (Ref. 4 Lists ± 2.5 deg). The pitch-roll variations listed in Table IV-B.1 suggest that atmospheric CO measurement data would be severely compromised due to Space Station attitude variations, if MicroMaps was rigidly fixed to the platform structure. Theoretically, the Space Station configuration will stabilize after operational capabilities are achieved, or funding evaporates, and the attitude variations listed in Table IV-B.1 will become fixed biases that could be corrected with counter bias mounting. However, the platform attitude control performance of ±3.5 to ±5 deg would still significantly contaminate the scientific data. Even in this scenario, angular variations will occur when the flight attitude mode is switched from the various options. To fully resolve these issues, MicroMaps would have to be mounted on an active pointing and/or tracking system. A pointing system with ±30 deg azimuth-elevation range could correct for the Table IV-B.1 and platform control performance variations in all flight attitude modes except XPOP. The XPOP mode demands a full 360 pointing capability (only 180 deg of usable pointing exists due to pallet viewing blockage).

As documented in Section II-B, scientific objectives emphasized precision nadir measurements of atmospheric CO vertical profiles. Mission scientists underscored this objective on several occasions. Therefore, based on platform suitability and associated attitude transient motions, flying MicroMaps on the International Space Station platform is not recommended, or would require an active pointing system. This option should be discarded.

C. Active Pointing System

If MicroMaps were to be flown on the International Space Station, an active pointing system would be required to counter platform dynamic motions, as discussed in the previous section. If this space basing option were selected, several candidate tracking system configurations would be available for mechanizing the MicroMaps pointing tasks. A Collocated Instrument-Tracker configuration is one example. In this arrangement, the science instrument is mounted on and is slewed with the tracking system in both azimuth and elevation axes. No reflective mirror is included with this arrangement. In contrast, a Separated Instrument-Tracker configuration could be utilized where the science instrument is physically separated from, and is not slewed with, the tracking system. In this arrangement, the tracking system consists of a gimbaled reflection mirror in both azimuth and elevation axes.

Figure IV-C.1 illustrates the more conservative collocated configuration. The main purpose of the tracking system is to provide stable, high accuracy pointing that counters undesirable platform transient angular motion. The system would consist of five major components with several subcomponents. Major components include the Base Unit, Lower Gimbal Unit, Upper Gimbal Unit, Control Unit, and Inertial Unit. The Base Unit is mounted directly to the Space Station pallet. The Lower Gimbal Unit is located on top of the Base Unit and is rotated relative to the Base Unit with the Azimuth Motor. The Azimuth Motor is housed internal to the Lower Gimbal Unit. The Upper Gimbal Unit is positioned on the top and side of the Lower Gimbal Unit and is rotated relative to the Lower Gimbal Unit with the Elevation Motor. The Elevation Motor is also housed internal to the Lower Gimbal Unit. The Upper Gimbal Unit contains the Science Instrument and Observation Camera. The reflected electromagnetic CO and visible spectrum signals enter the Science Instrument and Observation Camera through transparent windows provided on the Upper Gimbal Unit front surface. Finally, the Control Unit and Inertial Unit are also mounted directly to the Space Station pallet.

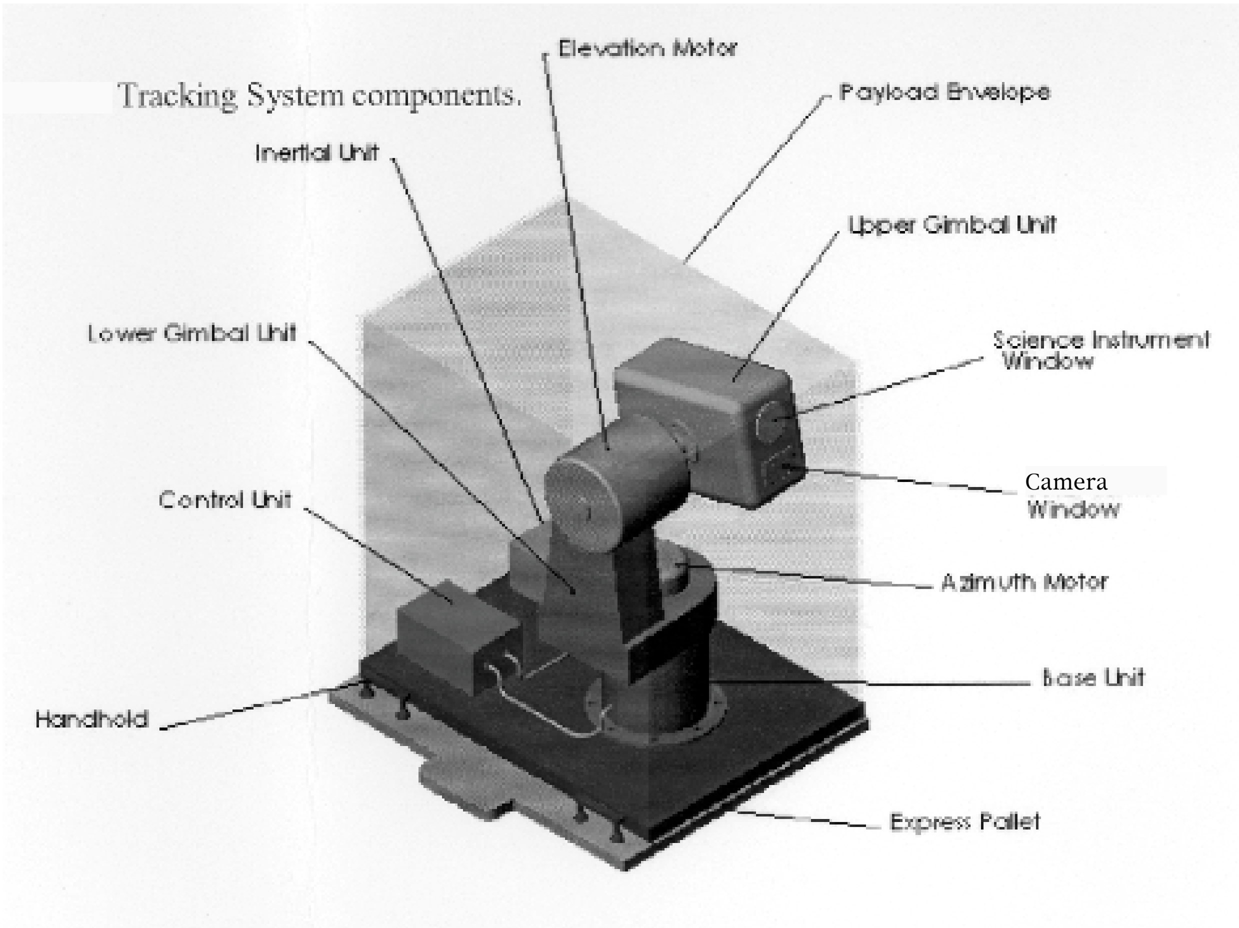

Figure IV-C.1 Collocated Instrument-Tracker Configuration

The Control Unit oversees and directs autonomously, or by up-link commands from mission control, all operations of the MicroMaps Tracking System. The Control Unit will activate and power-up the drive motors, sensors, and instrument subsystems. After system checks with all components and inertial reference solution convergence, the Control Unit issues open-loop actuation commands to the drive motors to slew the Science Instrument from its stowed position to the nadir vector computed from the inertial reference solution. The Control Unit then activates the closed-loop tracking mode to counter any platform disturbances. The inertial measurement signal is processed by the Control Unit control computer and Azimuth-Elevation Motor actuation commands are executed in order to hold the nadir pointing.

The MicroMaps Tracking System could employ high precision electric stepper motors coupled with harmonic drive mechanical transmissions, precise inertial sensing and estimation units, modern high throughput digital microprocessors to support hardware, and proportional-integral-derivative feedback compensation with lead biasing and cross axis feedforward logic. Such a system would provide robust operation with high levels of stability and performance that satisfy attitude pointing design requirements. However, this system would be costly and add significant complexity to the design system. If basing on the Space Station requires use of an active pointing system, then basing on a small dedicated spacecraft is of equivalent complexity. Flying MicroMaps on the International Space Station platform, although feasible, is still not recommended.

Section V
Launch Opportunities

A. Future Space Missions

Due to budget constraints, gaining access to the space environment for MicroMaps, regardless of the platform option selected, must be achieved by flying the instrument as a secondary payload on board an already scheduled flight. Domestic government and/or commercial launches to low Earth orbit with appropriate schedules and that satisfy the MicroMaps Mission requirements are highly desirable and sought. The initial task is simply to collect a database of future space missions form which appropriate launch opportunities can be identified. Using the internet and world wide web knowledge base is the most logical point to start the collection process. This search strategy proved to be very easy in identifying numerous lists of scheduled launches covering approximately 1 year from the present time. However, all of these flight lists are inconsistent with a MicroMaps Mission start date of 3-5 years from the present time. Consequently, a more refined search strategy is required. This strategy concentrated on identifying 1) multi decade space missions requiring multiple launches, 2) studies addressing demands for future space launch infrastructures, and 3) individual single launch space missions one by one, followed by focused searches on the identified items. This approach was successful in finding a large database of candidate missions.

Tables V-A.1 through V-A.3 list the results of this effort. Table V-A.1 lists many planned Space Shuttle and international launches from 1998 to 2006 which support construction and operation of the International Space Station.[21] If the International Space Station platform option was selected for MicroMaps, numerous Space Shuttle launch opportunities for access to low Earth orbit are noted in Table V-A.1. Table V-A.2 lists future, mostly domestic governmental and commercial science missions to be implemented out through the year 2020.[22] Many of these missions may offer suitable launch opportunities for a small dedicated spacecraft serving as the platform for MicroMaps. Finally, Table V-A.3 lists several future space missions currently under

Table V-A.1 Future International Space Station Support Missions

Schedule	Flight	Launcher	ISS Elements
Nov 1998	1A/R	Proton	FGB – Zarya
Dec 1998	2A	STS-88	Node 1- Unity: 1 Stowage rack, ZSRs, PMA1,PMA2; 2 APFRs (Sidewalls)
May 1999	2A.1	STS-96	Spacehab Double Cargo Module; OTD, Strela 1 Components, SHOSS (ICC)
May 2000	2A.2A	STS-101	Spacehab Double Cargo Module; Strela 1 Components, SHOSS (ICC)
Jul 2000	1R	Proton	Service Module (Zvezda)
Jul 2000	1P	Progress M-1	Standard Cargo
Sep 2000	2A.2B	STS-106	Spacehab Double Cargo Module; SHOSS (ICC)
Oct 2000	2R	Soyuz–TM(a)	ISS Crew (Expedition 1)
Nov 2000	2P	Progress M-1	Standard Cargo
Nov 2000	3A	STS-92	Z1 Truss Segment: CMGs, Ku-band, S-band Equip.; PMA3, 2 ETSDs (SLP); 2 Z1 DDCUs (Sidewall)
Nov 2000	4A	STS-97	PV Arrays (6 battery sets), EEATCS radiators, S-band Equip.
Jan 2001	5A	STS-98	U.S. Lab Module, Lab System racks, ZSRs; PDGF (Sidewall); SASA (Sidewall)
Feb 2001	3P	Progress M-1	Standard Cargo
Feb 2001	5A.1	STS-102	Lab System racks, RSRs, RSPs, ISPR (Lab Outfitting) (MPLM); EAS, PFCS, LCA, RU, ESP (ICC) (c)
Apr 2001	6A	STS-100	RSPs, RSRs, ISPRs (MPLM); DCSU (Sidewall); UHF, SSRMS (SLP) (b)
Apr 2001	2S	Soyuz-TM 32	ISS Crew (Taxi mission - Dennis Tito)
May 2001	7A	STS-104	Airlock: Stowage Platform, CA Equip Rack, Avionics Rack, Ext. Equip; High Pres. Gas Assembly (2 O2, 2 N2) (SLDP)
May 2001	4P	Progress M-1	Standard Cargo
Aug 2001	7A.1	STS-105	RSRs, RSPs, ISPRs (MPLM); SM MMOD Shields, SPP PWP Comp., OTD, 2 SHOSS, Ext. Att. P/L (ICC); APFR (Sidewalls) (c)
Aug 2001	5P	Progress M-1	Standard Cargo
Sep 2001	4R	Soyuz	Docking Compartment 1 (DC1): Strela 2
Oct 2001	3S	Soyuz-TMA	ISS Crew
Nov 2001	6P	Progress M-1	Standard Cargo
Nov 2001	UF1	STS-108	RSRs, RSPs, ISPR, MELFI (MPLM); WVS (Sidewall) (c)
Feb 2002	7P	Progress M-1	Standard Cargo
Feb 2002	9P	Progress M-1	Standard Cargo
Apr 2002	4S	Soyuz TM	ISS Crew
Apr 2002	8A	STS-110	S0: MT, GPS, Airlock and Node 3 Umbilicals, A/L, PWP
Apr 2002	10P	Progress M-1	Standard Cargo
May 2002	8P	Progress M	Standard Cargo
May 2002	UF2	STS-111	RSRs, RSPs, ISPRs (MPLM); MBS; PDGF (Sidewall); MDM Radiators (Sidewalls)
Jun 2002	11P	Progress M	Standard Cargo
Jun 2002	ULF1	STS-114	(MPLM); (ULC); ESP-2 w/ spares (c)
Jul 2002	9A	STS-112	S1: 3 TCS Radiators, CETA Cart A, S-band Equip.
Aug 2002	12P	Progress M-1	Standard Cargo
Oct 2002	11A	STS-115	P1: 3 TCS Radiators, CETA Cart B, UHF
Oct 2002	13P	Progress M-1	Standard Cargo
Oct 2002	9A.1	STS-116	Science Power Platform (SPP): 4 solar arrays, ERA, PDGF
Nov 2002	5S	Soyuz-TMA	ISS Crew
Dec 2002	14P	Progress M	Standard Cargo
Feb 2003	15P	Progress M-1	Standard Cargo
Feb 2003	12A	STS-118	P3; P4: PV Arrays (6 battery sets), 2 ULCAS
Mar 2003	12A.1	STS-119	Spacehab Single Cargo Module; (ICC);P5; PVRGF (Sidewall) (c)
Mar 2003	6S	Soyuz-TMA	ISS Crew
Apr 2003	16P	Progress M-1	Standard Cargo
Jun 2003	17P	Progress M-1	Standard Cargo
Jun 2003	13A	STS-121	S3; S4: PV Arrays (6 battery sets), 4 PAS

Schedule	Flight	Launcher	ISS Elements
Jul 2003	13A.1	STS-122	Spacehab Single Cargo Module; (ICC); S5; PVRGF (Sidewall)
Sep 2003	18P	Progress M-1	Standard Cargo
Sep 2003	7S	Soyuz-TMA	ISS Crew
Sep 2003	19P	Progress	Standard Cargo
Oct 2003	3R	TBD	Universal Docking Module (UDM)
Oct 2003	5R	TBD	Docking Compartment 2 (DC2)
Oct 2003	10A	STS-124	Node 2: DDCU racks, ZSRs; NTA (CBC)
Oct 2003	20P	Progress M	Standard Cargo
Jan 2004	1J/A	STS-125	JEM ELM PS: 4 Sys, 3 ISPRs, 1 Stow; 2 SPP SA w/truss, SM MMOD Shields (ULC); NTA; ORUs (CBC)
Feb 2004	ATV1	Ariane	TBD
Mar 2004	8S	Soyuz-TMA	ISS Crew
Apr 2004	9R	TBD	Docking & Stowage Module (DSM) [Spacehab/Energia Enterprise Module?]
Apr 2004	10A.1	STS-127	Propulsion Module
May 2004	1J	STS-126	JEM PM: 4 JEM Sys racks, JEM RMS
Jul 2004	21P	Progress M-1	Standard Cargo
Jul 2004	HTV	H-II DEMO	TBD
Aug 2004	UF3	STS-128	RSPs, RSRs, ISPRs, 1 JEM rack (MPLM); Express Pallet; ORUs (CBC)
Sep 2004	22P	Progress M	Standard Cargo
Sep 2004	UF4	STS-TB	S3 Attached P/L; ATA, SPDM (SLP)
Oct 2004	9S	Soyuz-TMA	ISS Crew
Nov 2004	23P	Progress M-1	Standard Cargo
Nov 2004	HTV1	H-II	TBD
Jan 2005	2J/A	STS-132	JEM EF; ELM-ES: EF Payloads, ICS, SFA w/carrier; Cupola (SLP); ATA
Feb 2005	24P	Progress M-1	Standard Cargo
Feb 2005	UF-5	STS-133	RSPs, RSRs, RSP-2s, ISPRs (MPLM); Express Pallet; ORUs (CBC)
Apr 2005	25P	Progress M	Standard Cargo
Apr 2005	10S	Soyuz-TMA	ISS Crew
May 2005	1E	STS-130	Columbus Module: ZSR, ISPRs; ORUs (CBC)
Jun 2005	26P	Progress M-1	Standard Cargo
Jun 2005	UF6	STS-136	RSPs, RSRs, RSP-2s, ISPRs (MPLM); Express Pallet; ORUs (CBC)
Jul 2005	14A	STS-TBD	2 SPP SAs w/truss, 4 SM MMOD Wings (ULC); MT/CETA Port & Stbd Rails (SLP), EF P/Ls & Spares
Aug 2005	27P	Progress M-1	Standard Cargo
Aug 2005	HTV2	H-II	TBD
Sep 2005	20A	STS-TBD	Node 3: Avionics Racks, ECLSS Racks; ORUs (CBC)
Sep 2005	16A	STS-138	Hab: Hab sys racks, ZSRs, 4 Crew Qtrs, ISPR; ORUs (CBC)
Oct 2005	28P	Progress M	Standard Cargo
Oct 2005	11S	Soyuz-TMA	ISS Crew
Nov 2005	8R	TBD	Research Module 1 (RM-1)
Dec 2005	17A	STS-139	1 Lab Sys, Node 3 System Racks, RSRs, RSPs, RSP-2s, ISPRs (MPLM); CBA (SLP) - (d)
Dec 2005	29P	Progress M-1	Standard Cargo
Jan 2006	18A	STS-140	CRV 1
Feb 2006	30P	Progress M-1	Standard Cargo
Feb 2006	19A	STS-141	RSPs, RSR, RSP-2s, ISPRs, Hab System Rack, CheCs Racks (MPLM); ORUs (CBC) - (e)
Feb 2006	HTV3	H-II	TBD
Mar 2006	31P	Progress M	Standard Cargo
Apr 2006	15A	STS-142	PV Arrays (6 battery sets), PV Module S6 Radiator
Apr 2006	12S	Soyuz-TMA	ISS Crew
May 2006	UF7	STS-143	Centrifuge Accommodations Module (CAM): ZSRs
May 2006	32P	Progress M-1	Standard Cargo
May 2006	10R	TBD	Research Module 2 (RM-2)
Jul 2006	33P	Progress M-1	Standard Cargo

Table V-A.2 Future Commercial and Government Science Missions

Schedule	Mission	Launcher
1998	Solcon-2	Heavy
1998	SLA-3	Heavy
1998	Defense Meteorological Satellite Program (DMSP-15)	Medium 1
1998	Mars 98 Orbiter	Medium 2
1999	Space Infrared Telescope Mount (SIRTM)	Heavy
1999	LANDSAT-7	Medium 1
1999	GOES-L	Medium 1
1999	Mars 98 Lander	Medium 2
1999	Stardust	Medium 2
1999	FUSE	Medium-Light
1999	HETE II	Ultra-Light
1999	SWAS	Ultra-Light
1999	WIRE	Ultra-Light
1999 Q1	HST Orbital Systems Test Platform (HOST)	STS
1999 Q1	International Extreme Ultraviolet Hitchhiker 03 (IEH-03)	STS
1999 Q1	Solar Extreme Ultraviolet Hitchhiker (SHE-03)	STS
1999 Q1	Ultraviolet Spectrograph Telescope for Astronomical Research (UVSTAR-03)	STS
1999 Q1	Spectrograph/Telescope for Astronomical Research (STAR-LITE)	STS
1999 Q1	Satelite de Aplicaciones-A (SAC-A)	STS
1999 Q2	Advanced X-Ray Astrophysics Facility (AXAF-1)	STS
1999 Q2	Southwest Ultraviolet Imaging System (SWUIS-02)	STS
2000	Defense Meteorological Satellite Program (DSMP-16)	Medium 1
2000	GOES-M	Medium 1
2000	Earth System Science Pathfinder (ESSP-1)	TBD
2000	TIMED-JASON	Medium 2
2000	Gravity Probe-B	Medium 2
2000	Image	Medium-Light
2000	MAP	Medium-Light
2000	HESSI	TBD
2000 Q3	Hubble Space Telescope Servicing Mission-03 (HST SM-03)	STS
2001	Space Readiness Coherent Lidar Experiment (SPARCLE, EO-2)	Heavy
2001	Defense Meteorological Satellite Program (DSMP-17)	Medium 1
2001	ICESAT	TBD
2001	Earth System Science Pathfinder (ESSP-2)	TBD
2001	Genesis	Medium 2
2001	Mars 01 Orbiter	Medium 2
2001	MIDEX	Medium-Light
2001	GALEX	TBD
2001	MARS 01 Lander	TBD
2001	SIRTF	TBD
2001-2003	UnESS (6 SC / yr)	Bantam
2002	Stratospheric Aerosol and Gas Experiment (SAGE III)	Heavy

Schedule	Mission	Launcher
2002	CIMEX	Heavy
2002	GOES-N	Medium 1
2002	CHEM-1	TBD
2002	SOLSTICE-SAVE	TBD
2002	Earth System Science Pathfinder (ESSP-3)	TBD
2002	Contour	Medium 2
2002	MIDEX	Medium-Light
2002	SMEX	TBD
2003	National Oceanographic and Atmospheric Administration (NOAA-N)	Medium 1
2003	Earth System Science Pathfinder (ESSP-4)	TBD
2003	EO3	TBD
2003	Europa Orbiter	STS
2003	DS-4	Medium 2
2003	MIDEX	Medium-Light
2003	SMEX	TBD
2003	DS-3 (3 SC)	TBD
2003	DS-5	TBD
2003	Europa Orbiter	TBD
2003	Mars 03 Lander	TBD
2003	Mars 03 Orbiter	TBD
2003	Discovery 7	TBD
2003 Q3	Hubble Space Telescope Servicing Mission-04 (HST SM-04)	STS
2004	Pluto-Kuiper Express	Medium 2
2004	MIDEX	Medium-Light
2004	SMEX	TBD
2004	STEREO (2 Launhes)	Ultra-Light
2004	Discovery 8	TBD
2004-2006	UNEX (6 SC / yr)	Bantam
2005	Advanced Cosmic Ray Composition Experiment for Space Station (ACCESS)	STS
2005	Mars Sample Return (2 SC)	Medium 3
2005	MS	Medium 3
2005	GLAST	Medium 3
2005	MIDEX	Medium-Light
2005	Magnet Multiscale (6 SC)	Medium-Light
2005	SMEX	TBD
2005	Discovery 9	TBD
2005	DS-6	TBD
2006	MIDEX	Medium-Light
2006	SMEX	TBD
2006	Discovery 10	TBD
2006	DS-7	TBD
2006	Mars 07 Lander	TBD
2007	Solar Probe	Medium 3
2007	MIDEX	Medium-Light
2007	Global Electrodynamics (GED, 5 SC)	Medium-Light
2007	SMEX	TBD

Schedule	Mission	Launcher
2007	Mars 07 Orbiter	TBD
2007	Discovery 11	TBD
2007	Next Generation Space Telescope (NGST)	TBD
2007	Solar-Mercury Mission	TBD
2008	ARISE	Medium 2
2008	MIDEX	Medium-Light
2008	Magnetospheric Constellation (MagCon, 20-100 SC)	Medium-Light
2008	SMEX	TBD
2008	Outer Planets 4	TBD
2008	Discovery 12	TBD
2008	DS-8	TBD
2009	MIDEX	Medium-Light
2009	SMEX	TBD
2009-2011	Constellation X (6 SC)	Medium 4
2009-2011	Solar Terrestrial Probe 7, 8 (2 SC)	Medium-Light
2009-2011	Mars Sample Return 09	TBD
2009-2011	Mars 11 Orbiter	TBD
2009-2011	Outer Planets 5 (2 SC)	TBD
2009-2011	Discovery 13-15 (3 SC)	TBD
2009-2011	DS-9	TBD
2009-2011	DS-10	TBD
2009-2011	Interstellar 1	TBD
2009-2011	Terrestrial Planet Finder (5 SC)	TBD
2009-2011	LISA (3 SC)	TBD
2010	MIDEX	Medium-Light
2010	SMEX	TBD
2011	MIDEX	Medium-Light
2011	SMEX	TBD
2012	MIDEX	Medium-Light
2012	SMEX	TBD
2012-2014	OWL (2 SC)	Medium 2
2012-2014	Solar Terrestrial Probe 9, 10 (2 SC)	Medium-Light
2012-2014	Mars (3 SC)	TBD
2012-2014	Outer Planets 6 (3 SC)	TBD
2012-2014	Discovery 16-19 (4 SC)	TBD
2012-2014	DS-11	TBD
2012-2014	DS-12	TBD
2012-2014	Interstellar 2	TBD
2013	MIDEX	Medium-Light
2013	SMEX	TBD
2014	MIDEX	Medium-Light
2014	SMEX	TBD
2015	MIDEX	Medium-Light
2015	SMEX	TBD
2015-2017	Solar Terrestrial Probe 11, 12 (2 SC)	Medium-Light
2015-2017	Mars (4 SC)	TBD

Schedule	Mission	Launcher
2015-2017	Outer Planets 7	TBD
2015-2017	Discovery 20-24 (5 SC)	TBD
2015-2017	DS-13	TBD
2015-2017	DS-14	TBD
2015-2017	Interstellar 3 (2 SC)	TBD
2016	MIDEX	Medium-Light
2016	SMEX	TBD
2017	MIDEX	Medium-Light
2017	SMEX	TBD
2018	MIDEX	Medium-Light
2018	SMEX	TBD
2018-2020	Solar Terrestrial Probe 13, 14 (2 SC)	Medium-Light
2018-2020	Mars (4 SC)	TBD
2018-2020	Outer Planets 8 (4 SC)	TBD
2018-2020	Discovery 25-30 (6 SC)	TBD
2018-2020	DS-15	TBD
2018-2020	DS-16	TBD
2018-2020	Planet Imager (21 SC)	TBD
2018-2020	Interstellar 4 (2 SC)	TBD
2019	MIDEX	Medium-Light
2019	SMEX	TBD
2020	MIDEX	Medium-Light
2020	SMEX	TBD

Heavy: 36,000 lbs to LEO
Medium 1: 10,000 lbs to GTO
Medium 2: 11,330 lbs to LEO (Delta II 5,139 kg)
Medium 3: 18,280 lbs to LEO (Delta III 8,292 kg)
Medium 4: 33,000 lbs to GTO (Delta IV 14,969 kg)
Ultra-Light: 4,012 lbs to LEO (Taurus 1,820 kg)
STS: Space Transportation System (Space Shuttle)

Table V-A.3 Individually Identified Future Space Missions

Schedule	Mission	Launcher
Jan 2003	ICESat (Ice, Cloud, and Land Elevation Satellite)	Delta II (7320)
Jan 2003	SORCE (Solar Radiation and Climate Experiment)	Pegasus XL
Jan 2003	CHIPS (Cosmic Hot Interstellar Plasma Spectrometer, UNEX, Launched With ICESat)	Delta II (7320)
Sep 2003	Swift (Gamma Ray Burst Explorer, MIDEX)	Delta II (7320)
Oct 2003	CINDI (Coupled Ion-Neutral Dynamics Investigation)	TBD
2004	FAME (Full-sky Astrometric Mapping Explorer, MIDEX)	Delta II (7425-10)
2004	TWINS (Two Wide-angle Imaging Neutral-atom Spectrometers)	TBD
2004	ACE (Atmospheric Climate Experiment)	Pegasus
Jan 2004	Aura (Coordinated With CALIPSO)	Delta II (7920-10L)
Apr 2004	CALIPSO (Cloud-Aerosol Lidar and Infrared Pathfinder Satellite Observations)	Delta II (7420-10)
Apr 2004	CloudSat (Launched With CALIPSO)	Delta II (7420-10)
Jun 2004	RADARSAT 2	Delta II
Jun 2004	POES (Polar Operational Environmental Satellites)	Delta II
Q3 2004	Discoverer II	Heavy-Medium
2005	SPIDR (Spectroscopy and Photometry of Intergalactic Diffuse Radiation, SMEX)	Pegasus XL
2005	POES (Polar Operational Environmental Satellites)	Soyuz (Russia)
Jan 2005	Astro-E2	TBD
Jul 2005	METOP (Meteorological Operational Polar satellites)	Atlas II / Ariane 5 (Europe)
Sep 2005	Solar-B	M-V (Japan)
Nov 2005	STEREO (Stereoscopic View of the Sun-Earth Connection)	TBD
2006	AIM (Aeronomy of Ice in the Mesosphere, SMEX)	TBD
2006	GOCE (Gravity Field and Steady-State Ocean Circulation Explorer)	Rockot (Russia)
2006	TechSAT-21	Delta IV-M
May 2006	NPP (NPOESS Preparatory Project)	TBD
Jun 2006	HYDROS	Taurus (2210)
Mar 2008	POES (Polar Operational Environmental Satellites)	Delta II
Feb 2009	MMS (Magnetospheric Multiscale)	TBD
Sep 2009	GEC (Geospace Electrodynamic Connections)	Delta
2010	POES (Polar Operational Environmental Satellites)	Soyuz (Russia)
Sep 2012	MC (Magnetospheric Constellation)	TBD
TBD	TRIANA	Space Shuttle
TBD	VCL (Vegetation Canopy Lidar)	TBD
TBD	OCO (Orbiting Carbon Observatory)	Taurus (2110)
TBD	AQUARIUS	TBD

development with a launch date well beyond the current time frame that were identified on a case by case basis. The associated launch opportunities may again be suitable for the MicroMaps Mission. Note these lists of future flight manifests are tentative and depend on factors such as future launch failures, program funding, political forces, and changing emphasis. However, these lists are appropriate for preliminary MicroMaps Mission studies.

B. Candidate Launch Assessments

Now that a healthy database of future space missions is available, the next task is to extract a subset of associated launch opportunities to low Earth orbit that are approximately in alignment with the MicroMaps Space Mission requirements. Top level (general) criteria such as launch year and orbit type can be used to narrow the database down to several competing launches. Then, a more refined assessment using additional lower level (specific) criteria such as launch date, orbital parameters, launch vehicle constraints, cost, and cooperation can be used to identify the optimum launch opportunity. This two stage approach will be used here. Before conducting the analysis, additional comments on this process are offered.

There are several considerations that must be made in order to launch a satellite as a secondary payload. The first consideration is the maximum payload capacity of the launcher and the mass of the primary payload and any additional secondary payloads previously scheduled. Next, the altitude must be approximately the same as desired. Now, this parameter is flexible since most geostationary orbiting satellites start off in low Earth orbit and are taken to geostationary Earth orbit by a separate booster rocket. However, inclination is different in that this is an inflexible parameter. Inclination changes are more difficult and expensive to make. Because satellites in geostationary Earth orbit have a low inclination, taking MicroMaps up as a secondary payload to a geostationary satellite is not appropriate to mission objectives. Of course, this is a conservative assessment of possible launch windows for MicroMaps. With an electric propulsion system, it would be possible to make changes in altitude and inclination. While it is desirable to find a launch with the correct orbit and inclination, there is some degree of freedom offered by an electric propulsion system.

Now return to the assessment task. In this analysis, the small dedicated spacecraft platform option for MicroMaps will be assumed. Further, recall the requirement for high orbital inclination from Section II-B. With this information, all Space Shuttle flights to the International Space Station in Table V-A.1 are eliminated from consideration. Further, any Space Shuttle flights listed in Table V-A.2 are also eliminated because they do not offer high orbital inclination. Now, recall from Section II the intended launch date for the MicroMaps Mission is 3-5 years off from the present time. Thus, only flights with a launch date lying approximately within the 2006-2008 window are retained. Additionally, only high inclination, low altitude flights are retained (which must be discerned from additional data not listed in Tables V-A.2 and V-A.3). The remaining launch opportunities that are potential contenders for the MicroMaps Mission are listed in Table V-B.1. The launch opportunities listed in this table are the results coming out of the initial assessment stage.

Table V-B.1 lists the high potential launch opportunities for MicroMaps with additional detail information on each mission including, orbital inclination, orbital altitude, mass constraints from the launch vehicle lift performance minus primary payload mass, size constraints from the launch vehicle fairing dimensions minus primary payload size, ascent constraints from the launch vehicle vibrational environment, cost sharing, and willingness for cooperation, where available. Data that is either not available, could not be found, or that must be collected in further studies, is designated as "To Be Determined". Because of the incomplete data, a final selection for the MicroMaps launch opportunity can not be made at this time. However, several important observations can be made and the steps necessary to complete this process at a later date are clear.

The missions METOP, Solar-B, GOCE, AIM, NNP, HYDROS, GED, POES, OCO, and AQUARIUS all have orbital geometries that can satisfy the MicroMaps Mission requirements. Orbital geometry parameters for the SPIDR mission are unknown at this time. However, with the

Table V-B.1 Potential Launch Opportunities for MicroMaps

Mission	Schedule (yr/mth)	Inclination (deg)	Altitude (km)	Mass Constraint (kg)	Size Constraint (m)	Ascent Constraint (g)	Cost Share ($)	Cooperation (-)
SPIDR	2005	TBD	TBD	Pegasus XL, Prime	Pegasus XL, Prime	Pegasus XL	≈ 0	High
METOP	2005 July	98.7, Sun Sync	796 – 844	Atlas II, Prime	Atlas II , Prime	Atlas II	TBD	TBD
Solar-B	2005 Sep	97.9, Sun Sync	600 km	M-V, Prime	M-V, Prime	M-V	TBD	TBD
GOCE	2006	96.5, Sun Sync	250	Rockot, Prime	Rockot, Prime	Rockot	TBD	TBD
AIM	2006	Polar Inclination	Low Altitude	TBD, Prime	TBD, Prime	TBD	≈ 0	High
MIDEX	2006	TBD	TBD	Medium-Light, Prime	Medium-Light, Prime	Medium-Light	≈ 0	High
SMEX	2006	TBD	TBD	TBD, Prime	TBD, Prime	TBD	≈ 0	High
NPP	2006 May	Polar Inclination	824	TBD, Prime	TBD, Prime	TBD	TBD	TBD
HYDROS	2006 Jun	Polar, Sun Sync	670	Taurus, Prime	Taurus, Prime	Taurus	TBD	TBD
GED	2007	Polar Inclination	350 – 2,000	Medium-Light, Prime	Medium-Light, Prime	Medium-Light	TBD	TBD
SMEX	2007	TBD	TBD	TBD, Prime	TBD, Prime	TBD	≈ 0	High
MIDEX	2008	TBD	TBD	Medium-Light, Prime	Medium-Light, Prime	Medium-Light	≈ 0	High
SMEX	2008	TBD	TBD	TBD, Prime	TBD, Prime	TBD	≈ 0	High
POES	2008 Mar	Polar Inclination	Low Altitude	Delta II, Prime	Delta II, Prime	Delta II	TBD	TBD
OCO	TBD	Polar Inclination	705	Taurus, Prime	Taurus, Prime	Taurus	TBD	TBD
AQUARIUS	TBD	Polar, Sun Sync	600	TBD, Prime	TBD, Prime	TBD	TBD	TBD

launch vehicle designated as the Pegasus XL, orbital altitude will be low and the inclination could be high. Thus, the SPIDR mission was retained in the final list. To discern between these missions, additional criteria must be considered. For example, higher altitude orbits could eliminate the need for a propulsion system and simplify the platform design. At this time, no attempt was made to quantify the constraints imposed by the launch vehicle, which could also expose the better opportunities. The AIM and SPIDR missions have been rated with minimal cost sharing and high cooperation because of their designation as low cost NASA Explorer Program missions (SMEX) which foster a spirit of cooperation in pursuing important but small scale scientific pursuits from space. In otherwords, an environment which facilitates secondary payloads to piggyback into space for minimal cost is present. The AIM mission may hold unique advantages in these latter criterion. This mission is being led by the Center for Atmospheric Sciences at Hampton University. The principal investigator is Dr. James Russell. The MicroMaps university team members and NASA Langley have a strong record of cooperation and close proximity with Hampton University and their atmospheric sciences program.

The various MIDEX and SMEX missions refer to the NASA Explorer Program flights slated for future launch, but have not been awarded to a specific proposal yet. The mission of the Explorer Program is to provide frequent flight opportunities for scientific investigations from space. The Explorer Program enables the definition, development and implementation of mission concepts through a variety of modes to meet the need of the scientific community and the NASA space science enterprise. The missions are characterized by relatively moderate cost, and by small to medium sized missions that are capable of being built, tested and launched in a short time interval compared to the large observatories. The three mission categories include Medium-class Explorers (MIDEX) where NASA expenses are not to exceed $150 M, Small Explorers (SMEX) where NASA expenses are not to exceed $75 M, and University-class Explorers (UNEX) where NASA expenses are not to exceed $15 M. Therefore, the generic MIDEX and SMEX launch opportunities listed in Table V-B.1 are projected to offer unique advantages, as well. When the MIDEX/SMEX awards are announced, their associated orbit requirements should be reviewed, and any that have been found consistent with the MicroMaps requirements, should be approached early on for future collaboration.

The launch opportunity for the MicroMaps Space Mission could very well come from this final list (Table V-B.1). After finding 16 strong possibilities in a preliminary study, securing a suitable launch for MicroMaps should be feasible. With additional information, possibly obtained from communicating with the mission lead personnel, the optimum launch opportunity can be identified. Another main point to make is that MicroMaps Mission planning and design should continue, so that when a launch opportunity presents itself, the MicroMaps team can quickly respond and take advantage of this opportunity. The MicroMaps team should be ready when these opportunities arise.

Section VI
Conclusions and Recommendations

A mission planning process was outlined and applied to specific aspects of the MicroMaps Space Mission. All constraint and objective information from various sources was quantified, documented, and mapped into requirements for orbital geometries and spacecraft subsystem characteristics. Further sizing and definition studies in these areas for a small dedicated spacecraft serving as the MicroMaps platform revealed no obvious critical requirements that would prevent a successful mission design and implementation. The most revealing result is an understanding of critical factors which impact the overall system design, and the key relationships between requirements, objectives, and constraints. Such understanding will be important when final engineering trades and program decision options are made. This study provides a framework that can be revisited when more detailed information is available in more advanced planning stages. The feasibility of using the International Space Station as a space platform for MicroMaps was evaluated in specific areas. Earth surface coverage, attitude and vibrational transients, and the need for an active pointing system revealed deficiencies with this space platform option. Some of these deficiencies could be overcome but with associated cost and complexity. Other deficiencies are simply not correctable. The secondary objective of enhancing and developing student skills in space systems would not be maximized with this option either. Based on these results, flying MicroMaps on the International Space Station is not recommended. A small dedicated spacecraft with a single function of supporting MicroMaps objectives is recommended. A large final list of launch opportunities with orbital characteristics and launch windows consistent with the MicroMaps Mission requirements was identified and described. Additional data and study will be needed to identify the optimum launch opportunity. The AIM mission, and future MIDEX/SMEX missions, offer unique advantages for MicroMaps.

Although a specific launch opportunity has not been recommended, results indicate finding such an opportunity should be solvable.

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(Acknowledgement and appreciation is given to Dr. Roscoe Lee, Northrop Grumman, and Dr. Khaled Abdel-Motagaly, Titan Systems, for providing References 17-19.)

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YYYY)*
01- 09 - 2003

2. REPORT TYPE
Contractor Report

3. DATES COVERED *(From - To)*

4. TITLE AND SUBTITLE
Engineering Feasibility and Trade Studies for the NASA/VSGC MicroMaps Space Mission

5a. CONTRACT NUMBER

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
Abdelkhalik, Ossama O.; Nairouz, Bassem; Weaver, Timothy; and Newman; Brett

5d. PROJECT NUMBER
L-16151

5e. TASK NUMBER

5f. WORK UNIT NUMBER
23-622-63-02

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
NASA Langley Research Center
Hampton, VA 23681-2199

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSOR/MONITOR'S ACRONYM(S)
NASA

11. SPONSOR/MONITOR'S REPORT NUMBER(S)
NASA/CR-2003-212652

12. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category 42
Availability: NASA CASI (301) 621-0390 Distribution: Standard

13. SUPPLEMENTARY NOTES
An electronic version can be found at http://techreports.larc.nasa.gov/ltrs/ or http://ntrs.nasa.gov
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14. ABSTRACT
Knowledge of airborne CO concentrations is critical for accurate scientific prediction of global scale atmospheric behavior. MicroMaps is an existing NASA owned gas filter radiometer instrument designed for space-based measurement of atmospheric CO vertical profiles. Due to programmatic changes, the instrument does not have access to the space environment and is in storage. MicroMaps hardware has significant potential for filling a critical scientific need, thus motivating concept studies for new and innovative scientific spaceflight missions that would leverage the MicroMaps heritage and investment, and contribute to new CO distribution data. This report describes engineering feasibility and trade studies for the NASA/VSGC MicroMaps Space Mission. Conceptual studies encompass 1) overall mission analysis and synthesis methodology, 2) major subsystem studies and detailed requirements development for an orbital platform option consisting of a small, single purpose spacecraft, 3) assessment of orbital platform option consisting of the International Space Station, and 4) survey of potential launch opportunities for gaining assess to orbit. Investigations are of a preliminary first-order nature. Results and recommendations from these activities are envisioned to support future MicroMaps Mission design decisions regarding program down select options leading to more advanced and mature phases.

15. SUBJECT TERMS
Carbon monoxide; Remote sensing; Mission design

16. SECURITY CLASSIFICATION OF: a. REPORT	b. ABSTRACT	c. THIS PAGE	17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON / 19b. TELEPHONE NUMBER *(Include area code)*
U	U	U	UU	101	STI Help Desk (email: help@sti.nasa.gov) (301) 621-0390